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WADC TECHNICAL NOTE 55-298

(UNCLASSIFIED TITLE)

WAR EMERGENCY THRUST AUGMENTATION
FOR THE J47 ENGINE IN THE F-86 AIRCRAFT

WILLIAM A. DAILEY, 1st Lt, USAF
POWER PLANT LABORATORY

AUGUST 1955

WRIGHT AIR DEVELOPMENT CENTER

JUN 5 1956

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**WAR EMERGENCY THRUST AUGMENTATION
FOR THE J47 ENGINE IN THE F-86 AIRCRAFT**

William A. Dailey, 1st Lt, USAF

Power Plant Laboratory

August 1955

Project No. T-11-P-206A-16

Wright Air Development Center
Air Research and Development Command
United States Air Force
Wright-Patterson Air Force Base, Ohio

JUN 15 1956

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FOREWORD

This document was prepared to serve as the final report of Power Plant Laboratory Project T-11-P-206A-16, (formerly S506-224J) entitled (Unclassified) "War Emergency Thrust Augmentation for the J47 Turbojet Engine Installed in the F-86 Aircraft". The project was administered by the Rotating Engine Branch, Power Plant Laboratory, Wright Air Development Center.

Acknowledgment for their work on the project is given to the following: F-86 Weapon System Project Office, Wright Air Development Center; Air Force Flight Test Center, Edwards Air Force Base; NACA Lewis Flight Propulsion Laboratory; General Electric Company; and North American Aviation Inc.

This document, excepting the title and Sections I and II, is classified CONFIDENTIAL because of the nature of, and potential future military application of the work reported on; classification is made in accordance with Air Force Regulation 205-1, paragraph 24a, dated 15 December 1953.

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ABSTRACT

Augmentating the thrust of the J47 engine in the F-86 aircraft was the principal objective of the program herein reported upon. The work covered a period of approximately two and one-half years and was conducted primarily by the General Electric Company and North American Aviation, Inc. under Air Force contract. Work was also accomplished on the project by the Wright Air Development Center, the Air Force Flight Test Center, and the Lewis Flight Propulsion Laboratory. All known schemes that could possibly augment the thrust of a turbojet engine were considered and overspeed, overtemperature, liquid nitrogen injection, water-alcohol injection, and pre-turbine fuel injection were brought under development. With the exception of overspeed and liquid nitrogen injection, the above systems were tested in flight and demonstrated that they could provide increased thrust for the J47 engine thereby substantially increasing the performance of the F-86 aircraft. Of the three augmentation systems flight tested, two drastically reduced the life of the engine and the third, water-alcohol injection, although not having such a severe effect on engine life was not suited for installation in the F-86 aircraft. Thus, under the circumstances, adaptation of any thrust augmentation system to the J47 engine in the F-86 aircraft for Korean operational use was deemed impractical.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

for C. C. Phillips
NORMAN C. APPOLD
Colonel, USAF
Chief, Power Plant Laboratory
Directorate of Laboratories

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TABLE OF CONTENTS

Section I	Overspeed	1
	A. General-Testing-Results	1
Section II	Overtemperature	3
	A. General	3
	B. Testing	4
	C. Results	6
Section III	Liquid Nitrogen Injection	13
	A. General	13
	B. Testing	14
	C. Results	17
Section IV	Water-Alcohol Injection	21
	A. General	21
	B. Testing	22
	C. Results	27
Section V	Pre-Turbine Injection	33
	A. General	33
	B. Testing	34
	C. Results	41
Section VI	General Conclusions	49
References		52
Appendix		54

CONFIDENTIAL

LIST OF ILLUSTRATIONS

<u>Figure</u>	<u>Page</u>
1. Project's Significant Events and Dates	x
2. J47-GE-13 Engine Thrust Vs RPM For Various Tail-Pipe Area Settings	2
3. J47-GE-13 Engine Compressor Efficiency Drop-Off With Increased RPM	2
4. F-86E Tail-Pipe Restrictor Segments Utilized In the J47-GE-13 Engine Overtemperature Tests	5
5. Auxiliary Throttle Stop Installed On the Throttle Quadrant In the Cockpit Of the F-86E Aircraft During the J47-GE-13 Engine Overtemperature Tests	5
6. Overtemperature Test Rate of Climb Data	7
7. Overtemperature Test Time to Climb Data	7
8. Overtemperature Test Level Flight True Air Speed Data	8
9. Overtemperature Test Engine Net Thrust Data	8
10. Overtemperature Test Specific Air Range Data	10
11. Overtemperature Test Engine Gross Thrust Data	10
12. Schematic Diagram Of the Liquid Nitrogen Test Set Up Utilizing the J47-GE-15 Engine	16
13. Liquid Nitrogen Injection Test Percent Augmentation Vs Injection Flow Rate	18
14. Liquid Nitrogen Injection Test Inlet Air Flow Temperature Decrease Vs Injection Flow Rate.	18

CONFIDENTIAL

15. Schematic Diagram Of the J47-GE-27 Engine Water-Alcohol Injection System Installed In the F-86F Aircraft	23
16. J47-GE-27 Engine Water-Alcohol Injection Control Panel On the Left Forward Console In the Cockpit of the F-86F Aircraft	25
17. F-86F Tail-Pipe Tab Assembly Utilized In the J47-GE-27 Engine Water-Alcohol Injection Testing	25
18. Water-Alcohol Injection Test Rate Of Climb Data	28
19. Water-Alcohol Injection Test Time To Climb Data	28
20. Water-Alcohol Injection Test Level Flight True Air Speed Data	30
21. Water-Alcohol Injection Test Engine Net Thrust Data	30
22. Water-Alcohol Injection Test Aircraft Acceleration Data	31
23. Water-Alcohol Injection Test Injection Flow Rate Data	31
24. Schematic Diagram Of the J47-GE-27 Pre-Turbine Fuel Injection System Installation In the F-86F Aircraft	36
25. F-86F Tail-Pipe Variable Area Nozzle Utilized In the J47-GE-27 Engine Pre-Turbine Fuel Injection Tests	39
26. J47-GE-27 Engine Pre-Turbine Fuel Injection Control Cockpit Presentation In the F-86F Aircraft	39
27. Pre-Turbine Fuel Injection Test Rate Of Climb Data	42

CONFIDENTIAL

28.	Pre-Turbine Fuel Injection Test Time To Climb Data	42
29.	Pre-Turbine Fuel Injection Test Level Flight True Air Speed Data	43
30.	Pre-Turbine Fuel Injection Test Engine Net Thrust Data	43
31.	Pre-Turbine Fuel Injection Test Aircraft Acceleration Time Data	44
32.	Pre-Turbine Fuel Injection Test Specific Fuel Consumption Data	444
33.	Pre-Turbine Fuel Injection Test Aircraft Load Factor Data	46
34.	Pre-Turbine Fuel Injection Test Radius Of Turn Data	46
35.	MIG-15 Rate Of Climb Data	54
36.	MIG-15 Time To Climb Data	54

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INTRODUCTION

Encounters against the enemy MiG-15 aircraft in Korea dictated that steps be taken to improve the performance of the F-86 aircraft. Combat experience demonstrated the inferiority of the combat and service ceilings of the F-86 aircraft and it was evident that the aircraft's climb performance was in need of the most improvement. Although the F-86 aircraft had higher level flight and dive speeds than the MiG-15 aircraft, the latter's ability to accelerate more rapidly to its maximum speed in part made up for its somewhat lower maximum attainable speed. It was directed that the greatest effort be exerted on measures to improve the combat capability of the F-86 aircraft with regard to the above mentioned aircraft performance variables. The product of these measures was to be something that could be placed in operational combat use at the earliest possible date; however, work was to be continued to provide further improvements as soon as they could be made available to supplant or supplement initial expediences adopted as emergency solutions. The understanding was that measures which promised significant performance gains would not be rejected on the grounds that engine life would be reduced unless the reduction in life imposed an unsupportable burden on supply and maintenance.

The deficiencies of the F-86 aircraft when compared to the MiG-15 aircraft, as outlined above, were indicative of one factor which needed improvement—thrust loading. In order to make the thrust loading of the F-86 aircraft more nearly equal to that MiG-15 aircraft, an improvement of such in the order of 50% would have been necessary. The problem was attacked by attempting to increase the thrust of the F-86 aircraft and reduce the aircraft's weight. All the known methods of augmenting the thrust of a turbojet engine were considered and many were actually brought under development for the J47 engine, the purpose being to improve the performance of the F-86 aircraft. This report will present and review the various technical aspects of the above cited problem which concern the J47 engine.

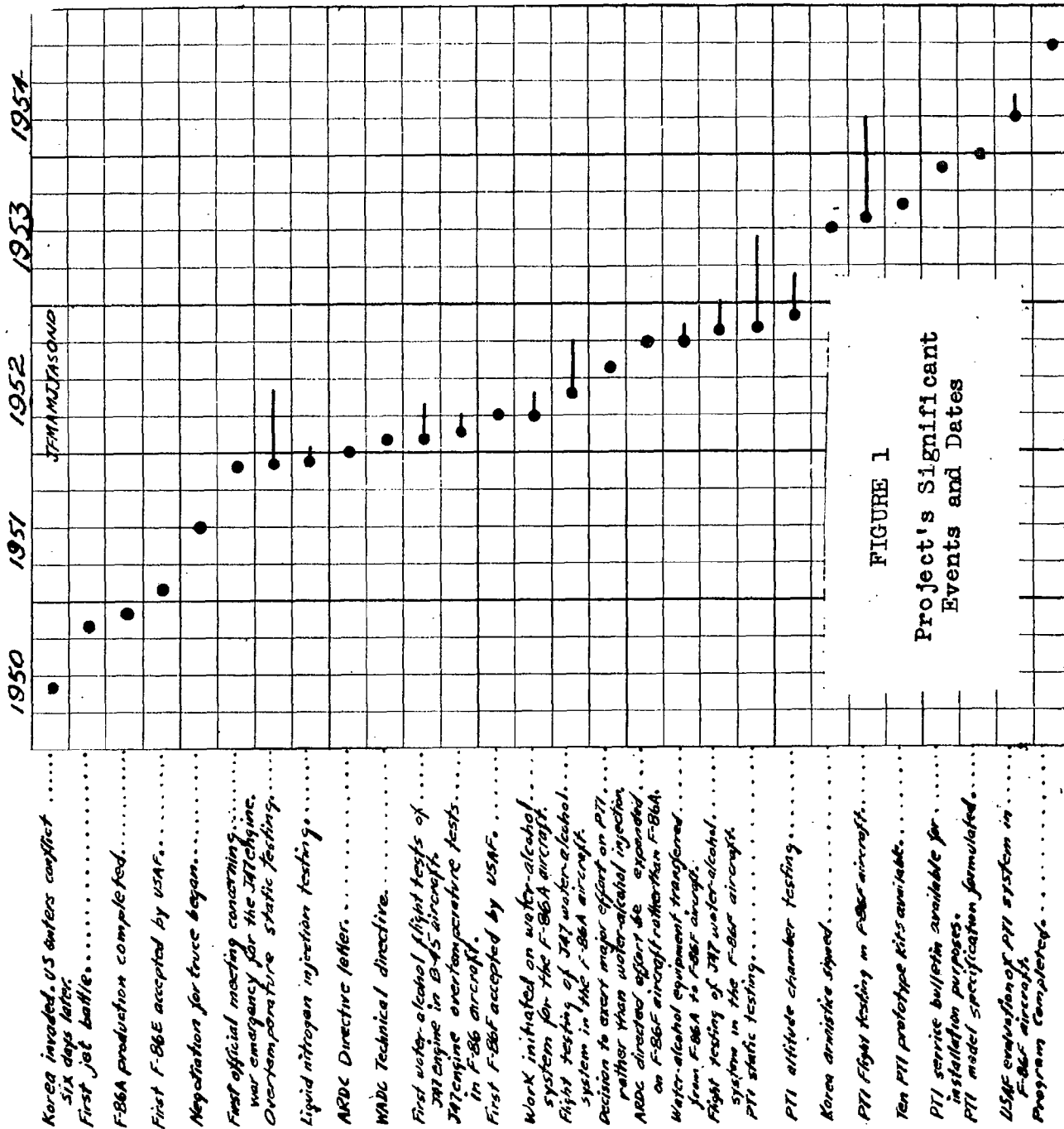
Because of the varied and extensive nature of the program concerning the efforts to augment the thrust of the

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J47 engine, the work reported upon herein is presented chronologically rather than in any order pertaining to the relative importance of the various phases of the work accomplished or the benefits derived from such work. As the object of the entire project was to meet an emergency, where timing was necessarily important, pertinent data with regard to this are presented in Figure 1 for early reference. Other steps taken to improve the performance of the F-86 aircraft, which were not directly related to the J47 engine, such as rocket boost, aerodynamic improvements, weight reductions, etc. are reviewed in numerous other USAF and contractors' documents and will not be subjects of this report.

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SECTION I

OVERSPEED

A. General - Testing - Results

Operating at a higher rotor speed was considered as a means of increasing the thrust of the J47-GE-13 engine; it was originally thought that an increase in thrust could be obtained by increasing the air mass flow through the engine. When the exhaust nozzle area was not enlarged for such operation, an overtemperature condition existed. However, it was soon found that when the exhaust nozzle area was so enlarged to maintain the limiting exhaust gas temperature, a decrease in thrust, rather than an increase occurred. Figure 2 illustrates how the thrust tapers off if the exhaust gas temperature is maintained constant by increasing the exhaust nozzle area for rotor speeds greater than 7950 rpm (100%). This condition was due to the comparatively large decrease in the pressure ratio across the nozzle relative to the increase in the air mass flow through the engine. Overspeed operation in itself contributed nothing to any additional thrust output when such operation was combined with overtemperature due to fall-off in compressor efficiency at the higher values of rpm; Figure 3 illustrates this fall-off in compressor efficiency. Should overspeed operation of the engine have resulted in an increase in thrust, it is doubtful if such operation could have been approved for any extensive service use because of the relatively large increase in centrifugal stresses which accompanied overspeed. These stresses increased approximately 8% when the engine was oversped by 4%.

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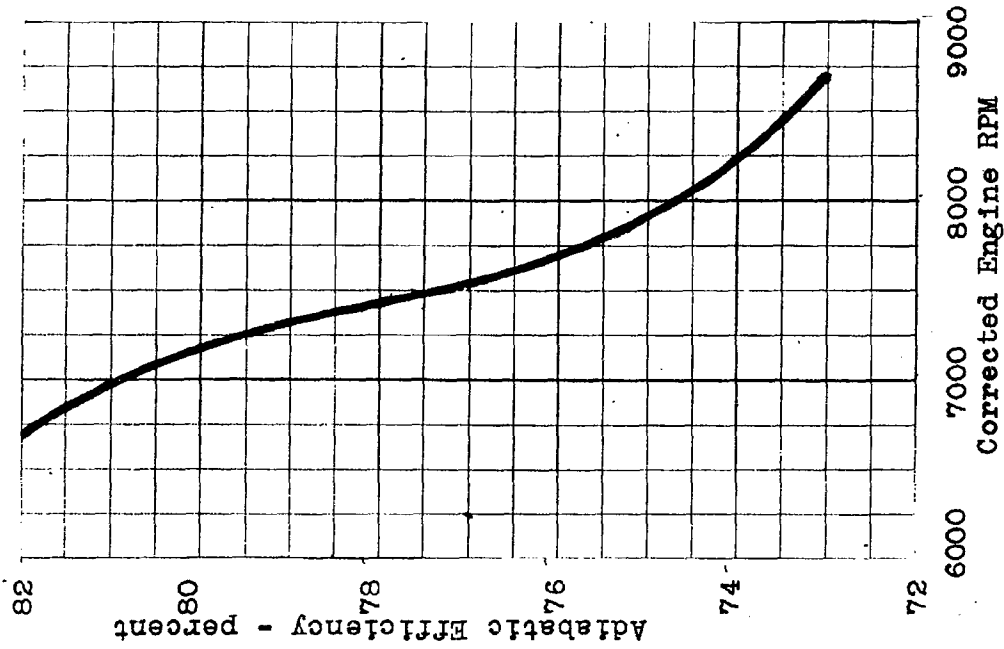


FIGURE 3. J47-GE-13 Compressor.

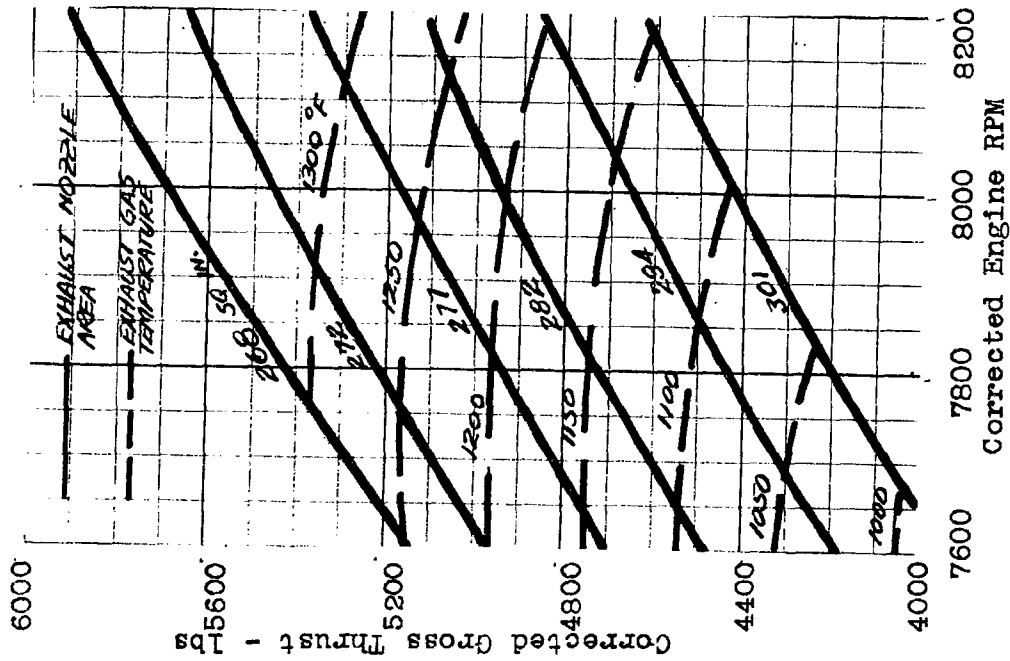


FIGURE 2. J47-GE-13 Engine.

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SECTION II

OVERTEMPERATURE

A. General

Previous experience had demonstrated that any thrust increase which was derived during the engine overspeed testing was solely due to overtemperature. Overtemperature operation, without overspeed, was then considered as a means of increasing the thrust of the J47-GE-13 engine. Although it was known that overtemperature operation would provide a substantial amount of augmentation, dependent upon the degree of overtemperature, there were many problems which had to be resolved before such operation could be approved for use in the field. The most difficult of these problems was that of determining a life for the engine parts that had been subjected to such overtemperature. It was known from past experience that realistic engine parts' life figures could only be obtained through service use or a long endurance testing program. Possible trouble areas which needed investigation were as follows: adequacy of aircraft cooling in parts adjacent to the engine, ability of the airframe structure to take the additional loading and temperatures, and the determining of accurate means of setting the rpm and exhaust nozzle area of the engine on the ground to give the desired conditions at altitude. Attention needed also to be given toward resolving the problems associated with the use of a smaller area exhaust nozzle with regard to engine control operation, engine acceleration, engine stalls, flame-outs, and starting.

Since it was not possible to enter into an endurance type of test program to gather engine parts' life data, it was decided to determine aircraft performance utilizing overtemperature operation of the engine and to gather as much other information as could be had, coincident with such performance testing, that might allow a reasonable assessment to be made about the degree of maintenance and logistic support which would be necessary to allow tactical use of engine overtemperature operation.

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B. Testing

Only a small expenditure of work was necessary to allow overtemperature operation and that consisted of placing restrictor segments in the tail-pipe to decrease the exhaust nozzle area. Figure 4 illustrates the restrictor segments placed in the tail-pipe of the J47-GE-13 engine in the F-86E test aircraft. In addition, an auxiliary stop was placed on the throttle quadrant in the cockpit. The auxiliary stop was installed on the throttle quadrant so that the engine might not be continually operated at an overtemperature condition. The auxiliary stop was set at a position which limited the rpm of the engine to a value such that 100% thrust was obtained at 100% exhaust gas temperature but at lower engine rpm than 100%. The auxiliary stop was designed so that the throttle could be pushed outboard and forward to the original full throttle position thus giving additional thrust resulting from overtemperature operation. Figure 5 illustrates the auxiliary stop installed on the throttle quadrant in the cockpit of the F-86E test aircraft. Such a system allowed normal operation up to the auxiliary stop and in addition allowed the pilot to obtain additional thrust by advancing the throttle past the auxiliary stop. Since the rpm was also increased when the throttle was advanced past the auxiliary stop, the air mass flow through the engine was increased slightly.

Approximately 99% thrust was available with the throttle at the auxiliary stop provided 100% exhaust gas temperature was maintained at that setting. Not only was 99% thrust available at 93% rpm, but due also to the engine's component characteristics, the full thrust normally available was obtained between 96% and 100% rpm at 100% exhaust gas temperature with due allowances being made for changes away from standard ambient conditions. Static and flight testing was accomplished with the auxiliary stop set at 7400 rpm (93%) and the exhaust nozzle area so reduced to produce 1275°F (100%) exhaust gas temperature at that engine rpm. With such an arrangement, an exhaust gas temperature of 1500°F (118%) was produced at 7950 rpm (100%). A timer was installed so that the duration of operation above 93% rpm could be determined. Flight operation above 93% rpm was for the most part limited to one minute cycles at the overtemperature setting in order to obtain maximum

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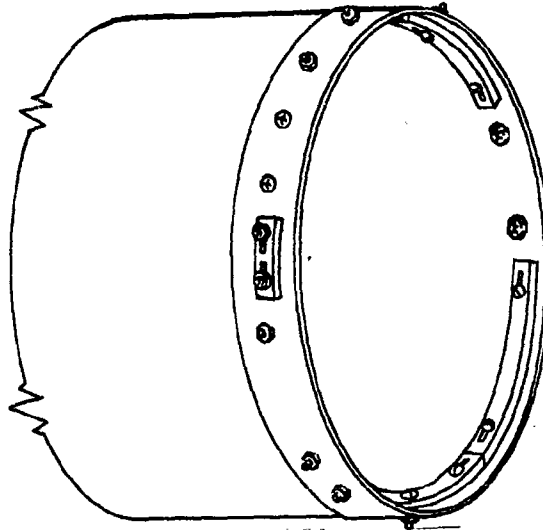


FIGURE 4. F-86E Tail-Pipe Restrictor Segments Utilized In The J47-GE-13 Engine Overtemperature Tests.

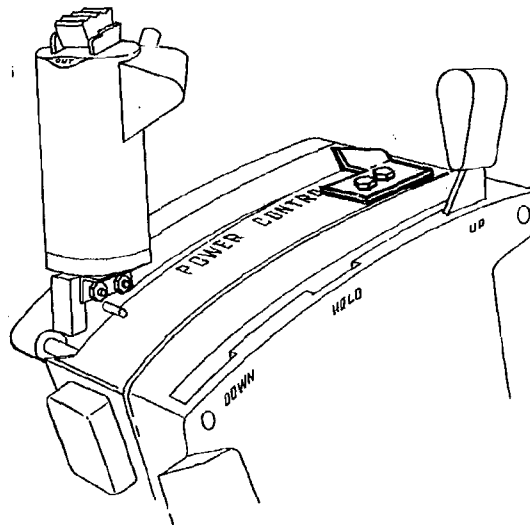


FIGURE 5. Auxiliary Throttle Stop Installed On The Throttle Quadrant In The Cockpit Of The F-86E Aircraft During The J47-GE-13 Engine Overtemperature Tests

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life of the turbine buckets. Continuous and cyclic overtemperature operation tests were conducted both on the ground and in flight. Throughout the testing, inspections were made in an attempt to correlate parts' life deterioration with overtemperature operation. A full metallurgical investigation of all parts subjected to overtemperature was made upon completion of the testing.

Two other tests were run in conjunction with the overtemperature flight testing. One such test was the placing of a by-pass needle valve between the large and small slot fuel manifolds to prevent exhaust gas temperature drop-off in climbs and the other was the placing of a 30 psi restrictor valve in the emergency fuel system side of the double check valve so as to give better temperature regulation while climbing. Both of these additional tests were conducted with the purpose of alleviating problems which had been experienced in the field.

C. Results

Figures 6 through 11 are plots of pertinent performance data gathered during the testing phase of an F-86E aircraft with a J47-GE-13 engine utilizing overtemperature as a means of thrust augmentation. Data for a standard unaugmented J47-GE-13 engine and F-86E aircraft are supplied for comparative purposes.

From Figure 6 it can be seen that the rate-of-climb of the F-86E aircraft was increased by 1800 ft/min at altitudes between 30,000 feet and 45,000 feet when utilizing overtemperature operation of the engine; thus the rate-of-climb was nearly doubled at an altitude of 35,000 feet and nearly tripled at an altitude of 40,000 feet. At an altitude of 45,000 feet the rate-of-climb with overtemperature operation was four times as great as the rate-of-climb at military power. The time to climb from an altitude of 30,000 feet to an altitude of 45,000 feet was reduced by approximately 8 minutes utilizing overtemperature operation of the engine as can be seen from Figure 7. Figure 8 shows a steady incremental increase in maximum level flight true air speed of about 10 knots at an altitude of 15,000 feet to 15 knots at an altitude of 45,000 feet utilizing overtemperature operation as compared to operation at mili-

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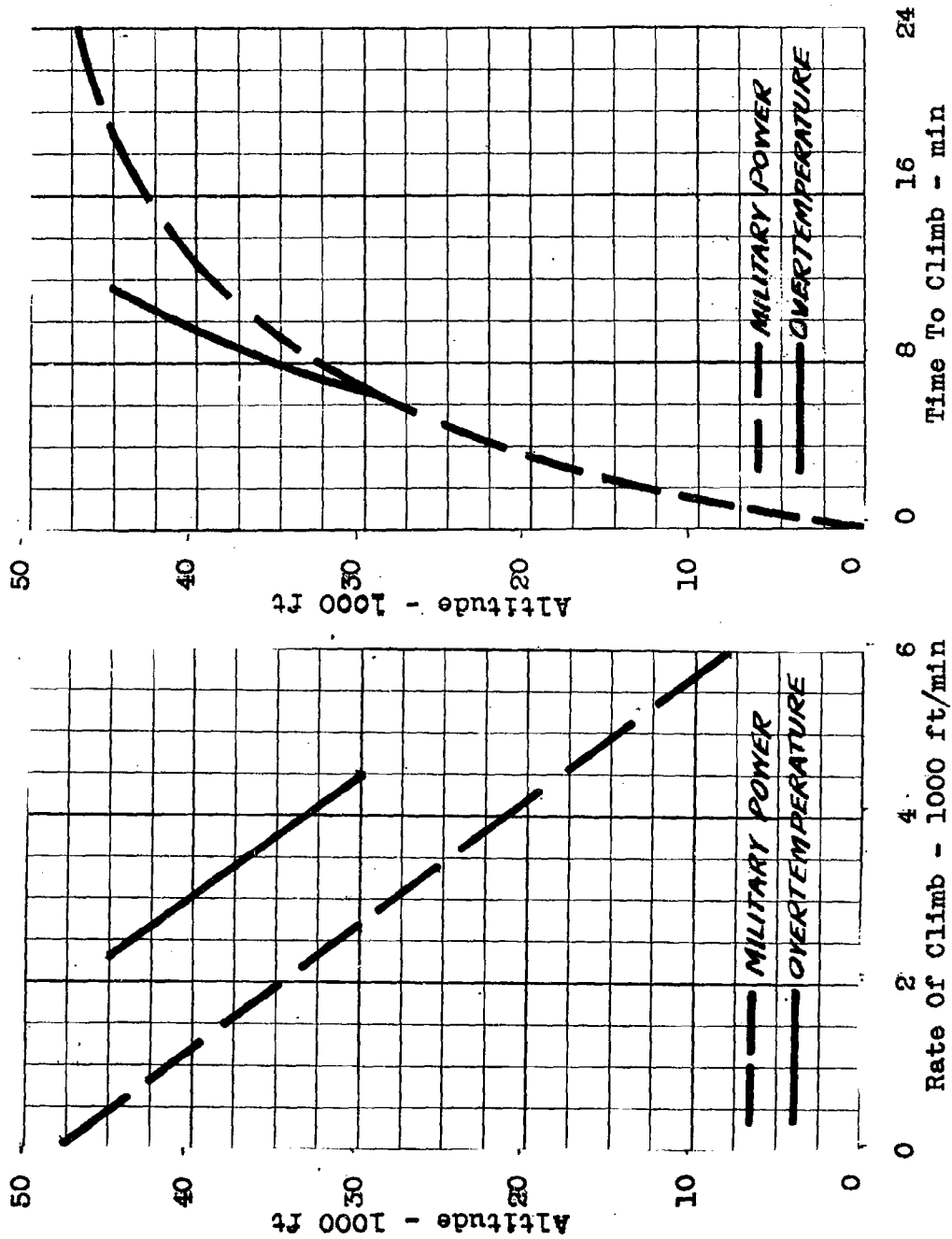


FIGURE 7. F-86E Aircraft. Combat Weight. Clean.

FIGURE 6. F-86E Aircraft. Combat Weight. Clean.

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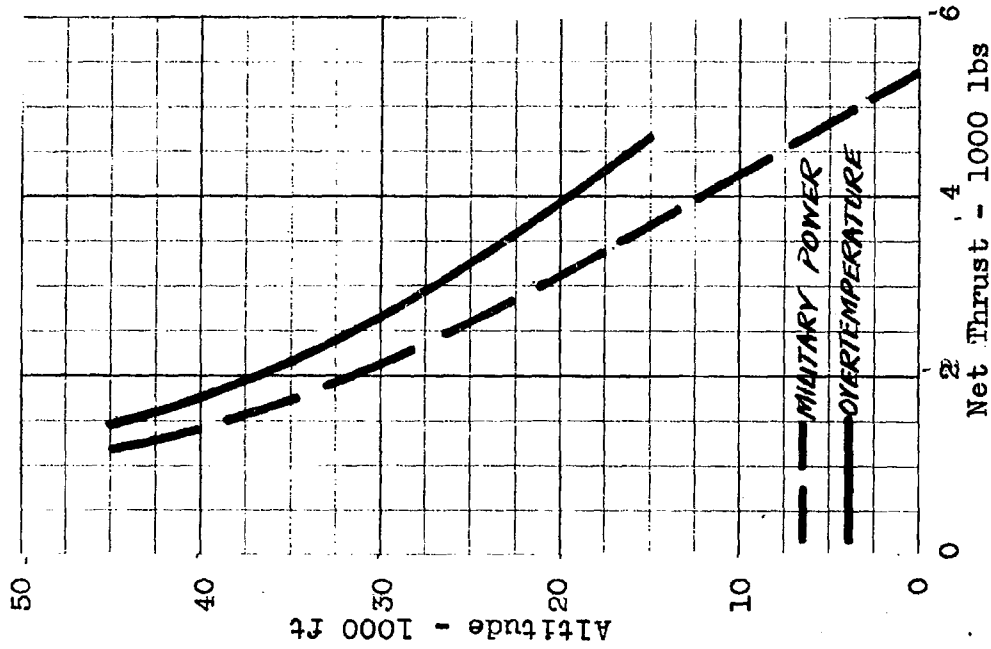


FIGURE 9. J47-GE-13 Engine.
F-86E Installation.

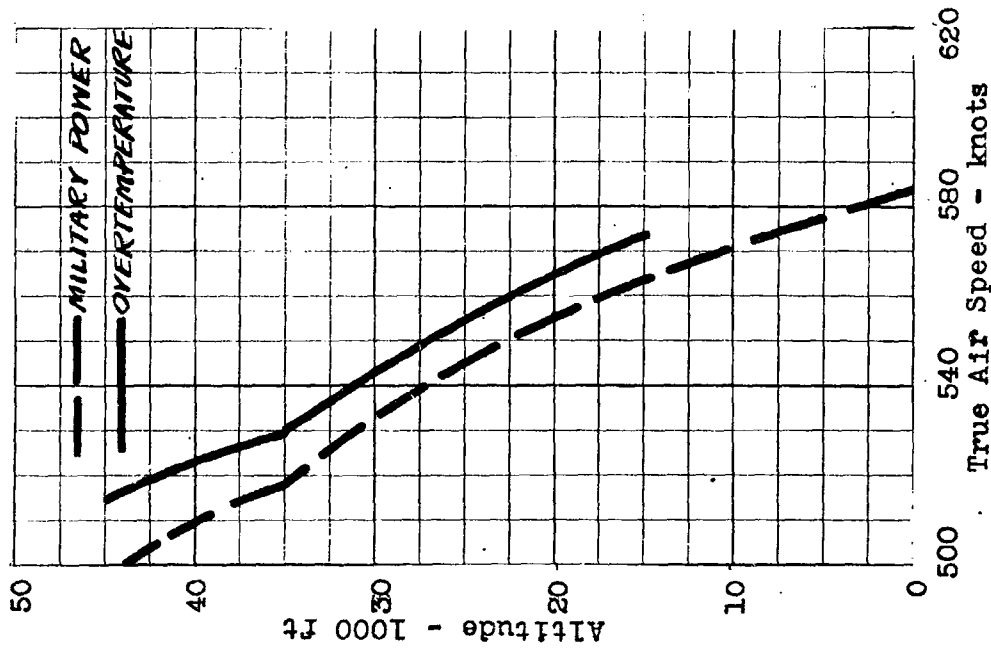


FIGURE 8. F-86E Aircraft. Combat Weight. Clean.

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tary power. The performance gains as noted were made possible by an average increase in net thrust of approximately 27% at altitudes between 15,000 and 45,000 feet, with a maximum increase in net thrust of nearly 30% being obtained at an altitude of 35,000 feet as can be observed from Figure 9.

As a result of the smaller exhaust nozzle area, a lower specific fuel consumption was obtained at the intermediate power settings within the cruise regime which was below 93% rpm. At the 100% rpm position, utilizing overtemperature operation, a somewhat lower or at least equal specific fuel consumption was realized. As detailed previously, no overtemperature operation was allowed below 93% rpm; 93% rpm was the point at which the auxiliary throttle stop was placed. Figure 10 shows a favorable increase in the range obtainable for the F-86E aircraft utilizing the smaller exhaust nozzle area necessary for overtemperature operation; the data indicates a 4 to 8% increase in range for a given fuel load, the cruising altitude naturally varying with aircraft weight. The curves presented in Figure 10 were obtained for best cruise speeds based upon calculation. Estimates indicated that an increase in combat radius of approximately 20 nautical miles was possible for the basic mission if flown with the non-standard exhaust nozzle area setting utilized for overtemperature operation.

As was stated previously, the deterioration of engine life was an important factor to be evaluated in the program. An accumulation of parts' life data allowed the following assessment of life to be given to each part: turbine buckets -- 10 minutes, transition liners, turbine nozzle diaphragm, and shroud ring -- 20 minutes, and inner combustion chambers -- 30 minutes. The above parts' life figures refer to overtemperature operation in a cumulative sense and represent the life of the parts for an increase in temperature of 225°F from the standard value. The exhaust-cone and tail-pipe were also adversely affected by the overtemperature operation; exhaust cone cracking was quite common. The results of the static tests showed that cyclic operation at overtemperature and continuous operation to the same total overtemperature level and duration were equally bad from a parts' life viewpoint.

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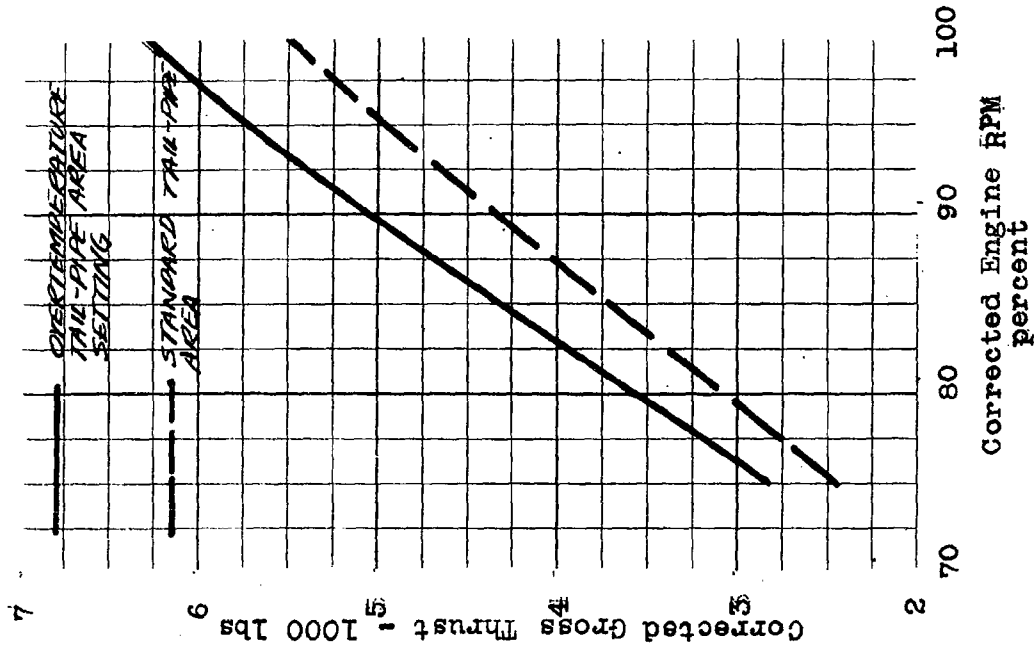


FIGURE 11. J47-GE-13 Engine.

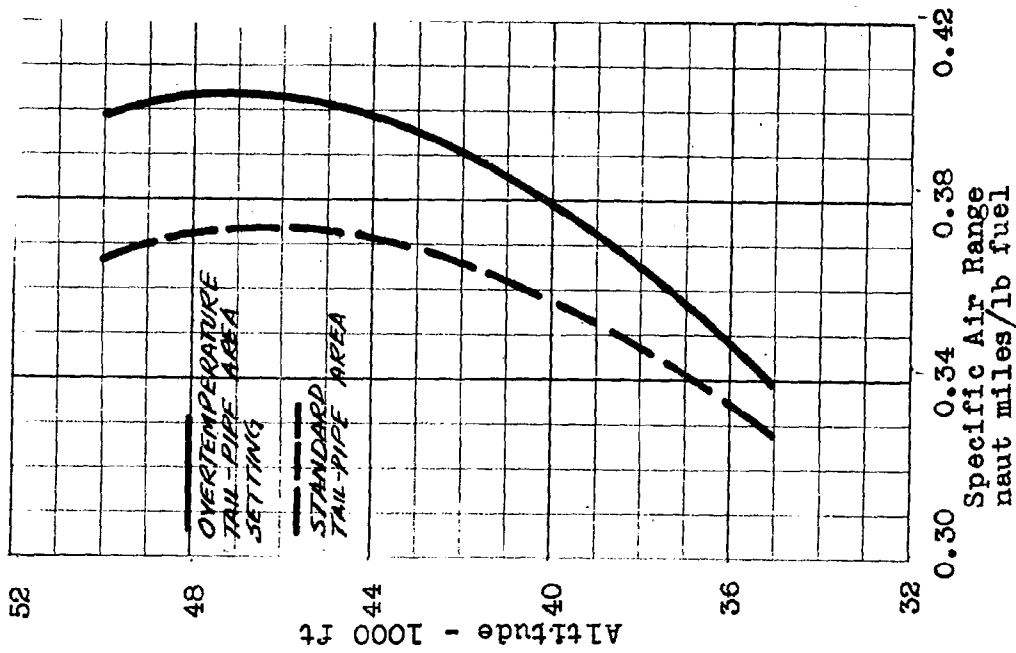


FIGURE 10. F-86E Aircraft.

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The ten minute life of the turbine buckets was found to be a limiting factor. In service use it would be necessary for all the buckets to be replaced at that time for it would not be possible to determine visually if the buckets were fit for further service. It was felt that the remaining parts of the engine subjected to overtemperature could be given a standard hot parts inspection and allowed to remain in service provided all the inspection criteria established in the already existing technical orders for such inspections were met. The effect of overtemperature on the turbine wheel was only determined to a very limited extent due to the difficulty surrounding such an investigation. In order to utilize overtemperature operation for the maximum time limit as determined by the life of the turbine buckets, it was necessary to begin the testing with new buckets. It was verified from the tests that if the overtemperature system as evaluated was to be used in the field with a reasonable factor of safety, new turbine buckets would have to be installed at the beginning of overtemperature operation and replaced after 10 minutes of overtemperature operation had been accomplished. Coincident with this turbine bucket replacement a thorough hot parts inspection was felt necessary. Such a bucket replacement and inspection task would necessitate removal of the aft fuselage of the aircraft and removal of the turbine wheel from the engine.

In general, the entire system functioned properly and no outstanding engine control or aircraft cooling problems were encountered. The test results indicated that the increase in performance of the F-86E aircraft resulting from overtemperature operation of the J47-GE-13 engine was substantial but the maintenance and logistic support for such operation made use of engine overtemperature operation impractical; thus, no overtemperature as such was utilized in Korea. A variation of the previous described technique used during the overtemperature tests was employed by one fighter wing in Korea, the purpose being to prevent exhaust gas temperature drop-off with altitude. The exhaust nozzle was tabbed by the placing of restrictor segments in the tail-pipe to produce rated temperature (100%) at 96% rpm while on the ground and as the temperature dropped off with altitude, the throttle was advanced by the pilot. Such a system if properly used did not overtemperature the engine, but it was necessary that the pilot closely monitor the

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exhaust gas temperature. The by-pass needle valve between the large and small slot fuel manifolds, which was evaluated during a portion of the overtemperature testing, was also used in Korea by another fighter wing to accomplish the same task of reducing exhaust gas temperature drop-off with altitude. Reduction in the temperature drop-off made possible the realization of more nearly the full available thrust at altitude and provided the limiting temperature was not exceeded, there would be no reduction in parts' life. As was previously stated, no loss in engine thrust occurred at the slightly reduced rpm, which was used for take-off and operation at the lower altitudes, provided 100% exhaust gas temperature was maintained.

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SECTION III

LIQUID NITROGEN INJECTION

A. General

One method of thrust augmentation for the J47 engine that appeared promising on the basis of analytical work conducted was that of liquid refrigerant injection into the compressor inlet. Such injection artificially cooled the entering air thereby resulting in a greater air mass flow through the engine and hence increased thrust. The lower temperature of the air flowing through the compressor increased the compressor pressure ratio by increasing the compressor Mach Number and also increased the difference between the temperature at which the work of compression was added to and taken from the working fluid in the engine. Liquid nitrogen and liquid oxygen were the refrigerants given the most consideration in the studies. It was concluded from a theoretical examination that for the same weight flows of refrigerant injection, liquid nitrogen and liquid oxygen would give approximately the same thrust augmentation providing the combustion process was not seriously disturbed by the injection of either refrigerant. Liquid oxygen is however more dense than liquid nitrogen and has the advantage of requiring a 14% smaller storage tank for the same weight. The use of liquid oxygen was later ruled out because it is dangerous from a handling point of view; it may explode spontaneously when brought in contact with grease or oil. Experience had also shown that leaks in installations utilizing liquid oxygen do occur and that the fire hazard is considerable; nitrogen on the other hand is inert and does not burn or explode.

Hydrogen peroxide, liquid ammonia, methyl chloride, and liquid air were also considered for possible use but soon eliminated from further consideration. Hydrogen peroxide was disqualified as a satisfactory coolant at the comparatively low operating temperatures since its boiling temperature is relatively high. Hydrogen peroxide also presented a distinct problem for it is spontaneously combustible at temperatures approximating the compressor discharge temperature of the J47 engine. Liquid ammonia,

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although possessing a boiling temperature sufficiently low to permit favorable evaporation at the temperatures and pressure considered is nevertheless toxic and since compressor discharge air was used for aircraft cabin pressurization, its use was ruled out. Liquid ammonia also attacks copper and copper alloys in the presence of moisture. Methyl chloride and liquid air were considered unsuitable because their heats of vaporization are comparatively low, and thus these coolants could absorb only small quantities of heat during vaporization. Water injection into the compressor inlet was unsatisfactory for use with the J47 engine due to the cooling of the compressor case and its subsequent contraction which caused interference between the rotor blades and the case. Naturally, cooling of the engine's compressor case would also occur with any type of refrigerant used, but it was thought that by injecting the coolant near the very beginning of the duct leading to the engine instead of directly at the compressor face, the throwing of the coolant outward toward the case as a result of centrifugal action could be avoided. It was doubtful if water injection into the compressor inlet would have been of much benefit at altitude because although water is a satisfactory coolant at the normal air temperatures associated with near sea level operation or for very high speeds at the higher altitudes, it is unsatisfactory at the low temperatures encountered by aircraft operating at moderate speeds at altitude since small amounts of water saturate the air and very little evaporate cooling can be obtained.

Since liquid oxygen appeared to offer no advantages as compared to liquid nitrogen injection, but instead offered many disadvantages in the elaborate care and precautions required for its safe handling and use, a test program was begun using liquid nitrogen. The purpose of initiating such a program was to determine whether satisfactory operation of the J47 engine could be maintained with liquid nitrogen when it was injected into the compressor inlet and also to determine the amount of thrust augmentation produced by such injection.

B. Testing

Only static engine testing was accomplished with fourteen actual test runs being made with liquid nitrogen in-

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jection. Figure 12 is a diagram of the test set up. Two tanks were used, one was a large supply tank and the other was a smaller pressure tank. To minimize heat transfer and the resulting loss of liquid nitrogen, the two tanks and the related plumbing were wrapped with an insulating foil. Gaseous nitrogen was used to pressurize the liquid nitrogen supply tank to force the liquid through the injection nozzle. A manually controlled pressure regulator was used to control the flow of gaseous nitrogen into the liquid nitrogen tank and the resulting liquid through the injection nozzle. An F-86 aircraft duct was placed ahead of the engine and was separated from it by a plenum chamber. The injection nozzle was located at the very beginning of the duct and extending directly into the center, being supported by its single supply line which extended out from one side of the duct. The position of the liquid nitrogen injection nozzle was approximately 24 feet linear horizontal distance from the plane of the engine compressor inlet. The nozzle stem axis was parallel to the axis of the engine. The geometry of the nozzle was such that the nitrogen was injected in a continuous sheet which in still air would form a cone with an included angle of approximately 100 degrees. The geometric apex of the cone pointed downstream so that there was an upstream component of injection velocity. However, when the engine was running at near full power, the velocity of the entering air overcame the upstream component of the nitrogen injection velocity so that the injected sheet of liquid nitrogen took on the appearance of a paraboloid of revolution with its vertex located at the injection nozzle with the concave formation opening downstream. Originally it was felt that the smallest practical nozzle orifice size and the largest practical nozzle injection pressure would provide the optimum vaporization because such a combination would provide the best atomization. Optimum vaporization was desired because if the liquid nitrogen was not completely vaporized before reaching the compressor inlet, a portion of the potential cooling effect on the engine airflow, and consequently maximum thrust augmentation would not be realized. It was later found that another factor influenced the amount of vaporization. It appeared that with a greater nozzle opening, larger slugs of liquid nitrogen were injected into the engine airflow and it was reasoned that the increased mass and added momentum were such that it resulted in the

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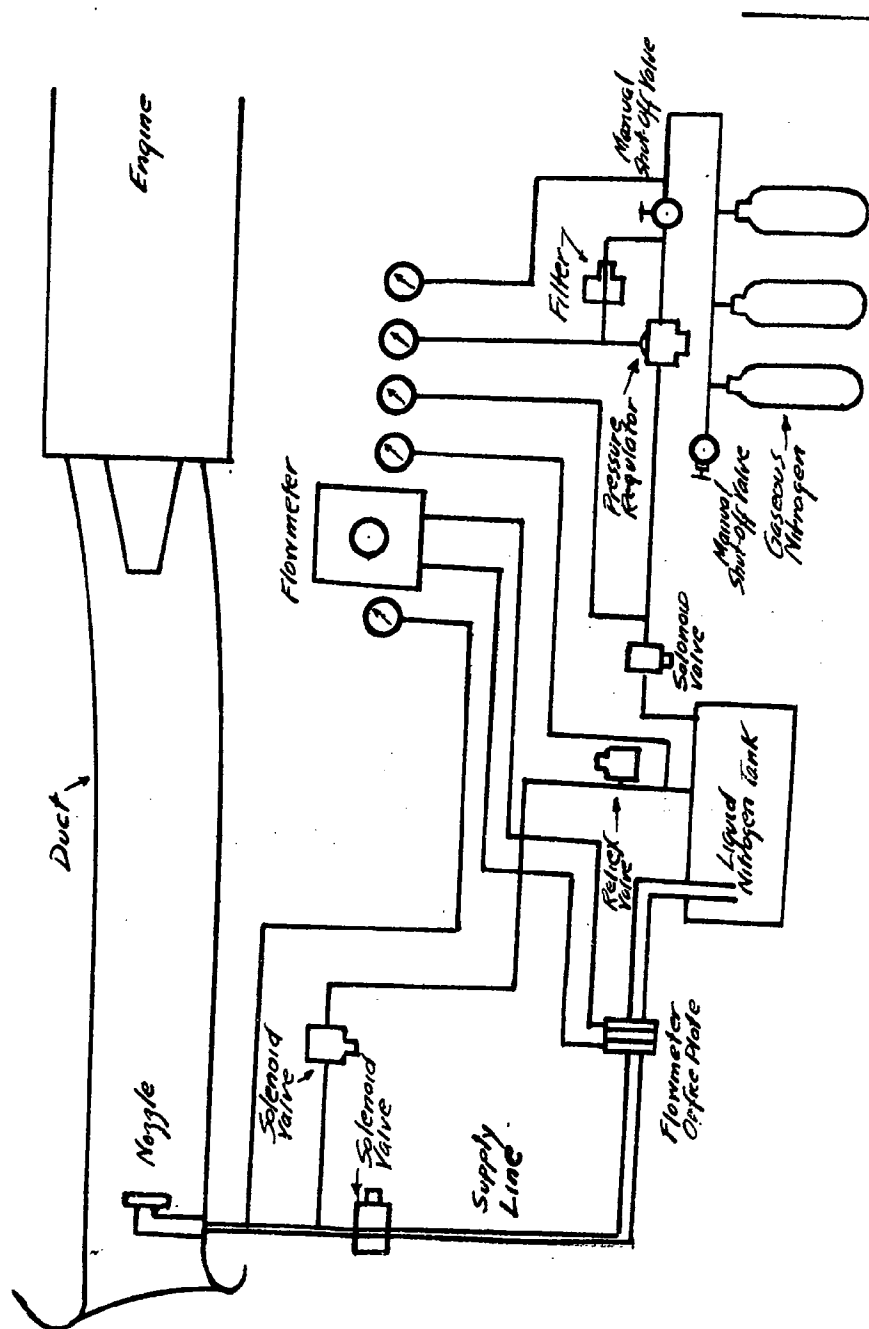


FIGURE 12. Schematic Diagram Of The Liquid Nitrogen Test Set Up Utilizing The J47-GE-15 Engine.

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liquid nitrogen penetrating the engine airflow all the way to the duct walls where the nitrogen was spread out in a thin film and vaporized by the agitating nature of the boundary layer. Thus better results were obtained by using a larger nozzle opening than was originally thought necessary.

To offset the exhaust gas temperature drop-off during liquid nitrogen injection, the same method previously described was utilized; thus prior to the test, the exhaust nozzle area was so adjusted to produce the maximum allowable continuous exhaust gas temperature at approximately 93% rpm. The throttle was then advanced during injection, usually to approximately 97% rpm in order to maintain the exhaust gas temperature near its maximum allowable limit.

C. Results

Figure 13 shows the percent thrust augmentation obtained for various liquid nitrogen injection rates utilizing the J47 engine in static tests. It can be seen that at the relatively high liquid nitrogen injection rates in the order of 17 lbs/sec an increase in thrust augmentation of approximately 28% was obtained. There was quite a discrepancy between the data obtained from preliminary calculation and the actual test results; it was believed to be caused by the poor vaporizing ability of the injection nozzle. Thus a smaller increase in actual inlet weight flow and compressor pressure ratio were obtained which resulted in a lower thrust due to the non-uniform inlet-air temperature distribution. It appeared that the larger the nozzle openings and the higher the injection rates, the worst the engine inlet temperature distributions were. As was previously pointed out, the utilization of a somewhat larger nozzle than was originally anticipated had also a rather beneficial effect.

Figure 14 shows the approximate maximum temperature decrease of the airflow entering the engine as a result of liquid nitrogen injection. The data presented in the figure were obtained after the injection flow had built up to a relatively constant value. The approximate point to point temperature reduction of the airflow across the compressor inlet showed a $\pm 10\%$ to $\pm 14\%$ variation from the mean value with liquid nitrogen injection. The total pressure loss of the incoming air to the engine due to the injection of

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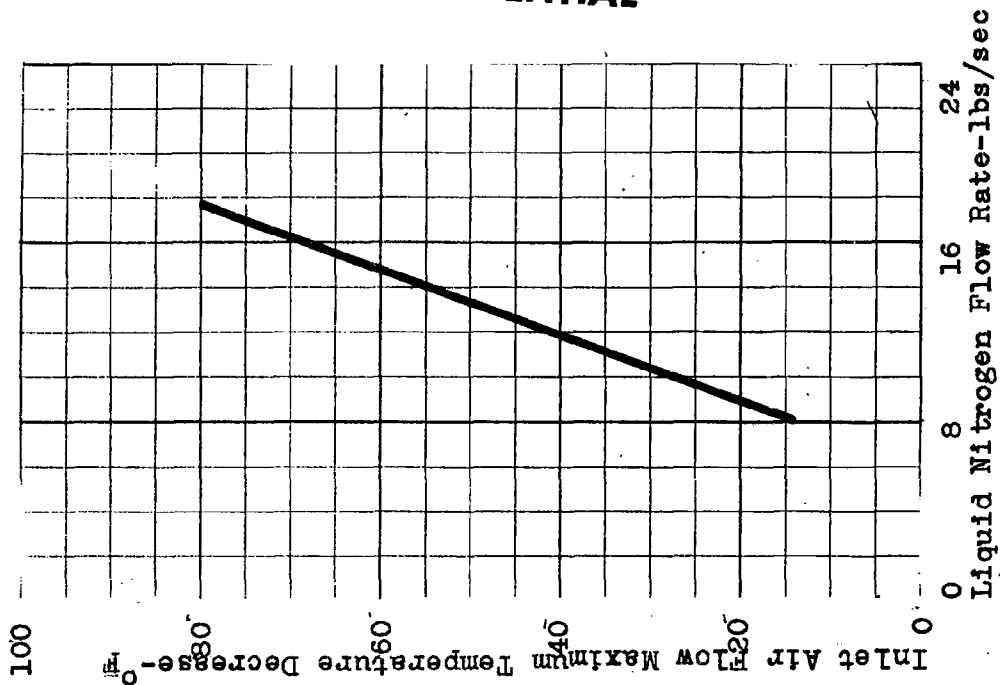


FIGURE 14. J47-GE-15 Engine.

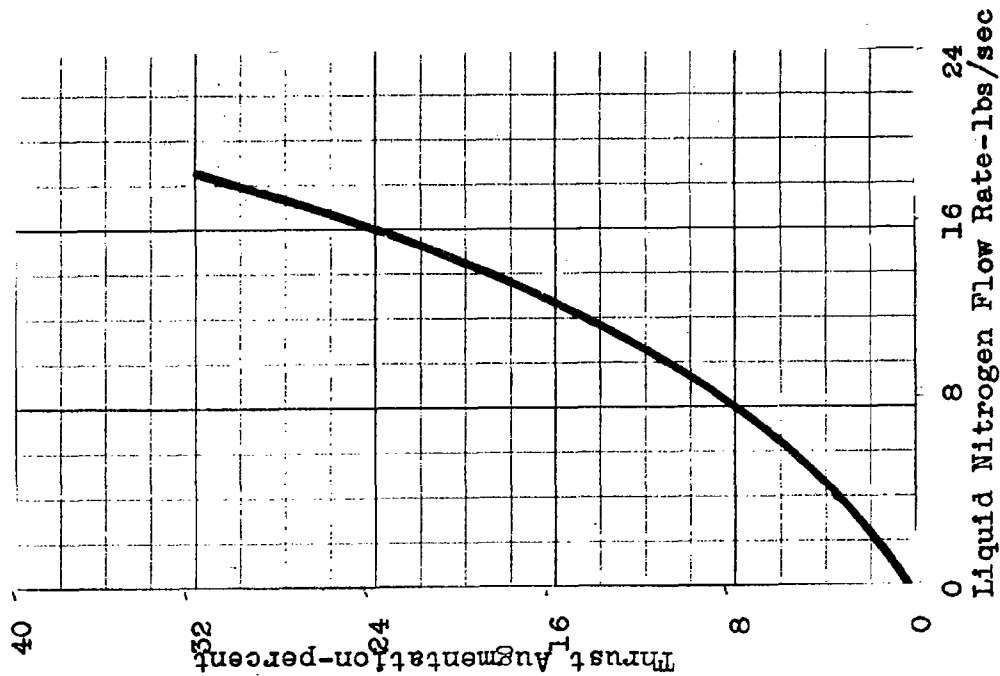


FIGURE 13. J47-GE-15 Engine.

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liquid nitrogen was about twice as much as was indicated by preliminary calculation. It was felt that the higher than anticipated pressure loss could be attributed to the fact that the nitrogen was injected at an angle having an upstream velocity component. A loss in free stream total pressure of 2% due to nitrogen injection would, by calculation, amount to a net thrust decrement of about 115 lbs for the J47-GE-13 engine in the F-86 aircraft at an altitude of 30,000 feet and a true air speed of 550 knots. Thus pressure loss was a more important factor than theoretical analysis had indicated. However, it is felt that additional nozzle development could minimize the total pressure loss due to injection. Engine combustion failure or flame-out occurred in 40% of the tests under similar conditions. It is believed that they were precipitated by excessive rates of liquid nitrogen injection in excess of 17 lbs/sec. It is significant then that the flame-outs occurred when the compressor inlet total temperature was in the region of 400°R.

In handling the liquid nitrogen during the tests, the fact became generally established that liquid nitrogen was not nearly as volatile as some references had pointed out. The nitrogen tanks were prone to leaking at any place there was a bolt in a hole due to the cooling effect on the metals and the differential contraction between the two. Although the handling of the liquid nitrogen appeared quite reasonable, its storage could prove quite difficult and its availability might be limited as a result; another factor was the excessive weight and space necessary for insulating aircraft storage tanks. No data was collected on the effect that liquid nitrogen injection into the engine would have upon the cabin pressurization equipment since no attempt was made to adapt the system to an F-86 aircraft. It was thought that through continual development of the system, a substantial increase in thrust could be gained even at the higher altitudes; but, it was also thought that the limitations imposed on the system from an aircraft modification and weight standpoint made it impractical. There was also the time that had to be made available to acquire flight test data on such a system and it was believed that concentration should be centered on a system that showed promise of being more easily adapted.

Liquid nitrogen injection was eliminated from further consideration as a means of augmenting the thrust of the

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J47 engine in the F-86 aircraft even though static testing and estimated performance at altitude had showed promise. It was nevertheless true that the full potential of such a system could only be realized at relatively high ambient temperatures or very high speed operation and such which would not be the case for operation in Korea with the F-86 aircraft against the enemy MIG-15 aircraft.

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SECTION IV

WATER-ALCOHOL INJECTION

A. General

Although analytical work showed that liquid injection into the compressor might prove highly satisfactory, such operation with the J47 engine was not possible due to the cooling and subsequent contraction of the compressor case causing interference between the case and the rotor blades; therefore, the only alternative was to inject directly into the combustion chambers. Work on a water-alcohol combustion chamber injection system for the J47-GE-13 engine in the F-86 aircraft was therefore initiated. The basic idea behind the water-alcohol combustion chamber injection system as applied to the J47 engine was that by virtue of the liquid injection, the fluid weight flow through the engine could be increased. In addition, the exhaust gas pressure would also increase. Both of these factors allowed increased thrust. It was necessary to mix alcohol with the water so as to supply the heat required for vaporization of the water. Some work on such a system had already been accomplished with the J47 engine prior to the initiation of a formal program to meet the then present emergency, but it was confined to static testing.

There were many problems associated with the use of a water-alcohol combustion chamber injection system in the F-86 aircraft that had to be investigated. Also, it was not known prior to the initiation of flight testing just exactly what increase in aircraft performance might be realized with such a system. The problem of making a mechanically satisfactory water-alcohol injection installation in the F-86 aircraft was a difficult one since space was extremely limited. The actual components to be used in the aircraft portion of the system presented problems for none were specifically designed for such an installation. In the interest of safety, the preliminary tests using water-alcohol injection were made on one engine of a B-45 aircraft. Later testing was accomplished with a J47-GE-13 engine in a F-86A aircraft. Final testing was accomplished with a J47-GE-27 engine in a F-86F aircraft for the F-86E aircraft was scheduled to be phased out of combat.

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B. Testing

The water-alcohol mixtures utilized in the testing consisted of plain tap water, AN-A-18 alcohol (MIL-A-6091), and emulsive corrosion preventive oil, USAF Specification 3604-A. The mixtures were prepared on a volumetric basis. At first, the mixtures were prepared by mixing the oil and water, then adding this mixture to the alcohol. Later, better results were obtained by adding the water to the alcohol and mixing to the desired percentage, then adding the oil. Figure 15 is a diagram of the water-alcohol injection system tested in the F-86 aircraft. The aircraft's aft fuselage 105 gallon fuel tank was isolated for use as a water-alcohol tank. On the original installation the normal tank outlet line was used as a water-alcohol supply line and the fuel transfer pump was replaced by a pump which had been modified for use as a water-alcohol boost pump. It was soon learned that the tank outlet line offered too much restriction and the boost pump could not supply the flow required to keep the injection system in operation at the lower altitudes, so the tank outlet line was capped and use of the boost pump was abandoned.

For the next configuration a plate was made to fit in place of the fuel level transmitter on top of the tank and a 1-3/4 inch diameter tube was welded to the plate and formed in such a way that it extended downward to within an inch of the bottom of the tank. Air for tank pressurization was obtained from the line used for pressurizing the main hydraulic reservoir. A one-half inch diameter line was installed between the tee downstream from the aircraft's primary heat exchanger and one of the tank vent lines. A gate type shut-off valve was installed in the line, and a relief valve capable of passing high airflows was used to limit maximum tank pressure to 8 psi. The other tank vent lines were capped. This configuration proved very successful and was used for the remainder of the test on the F-86A aircraft.

For the test on the F-86A aircraft, the J47-GE-13 engine was equipped with thimble type combustion chamber liners since it had been determined that the combustion characteristics, when utilizing these liners, were better when used in place of the standard liners. For approxi-

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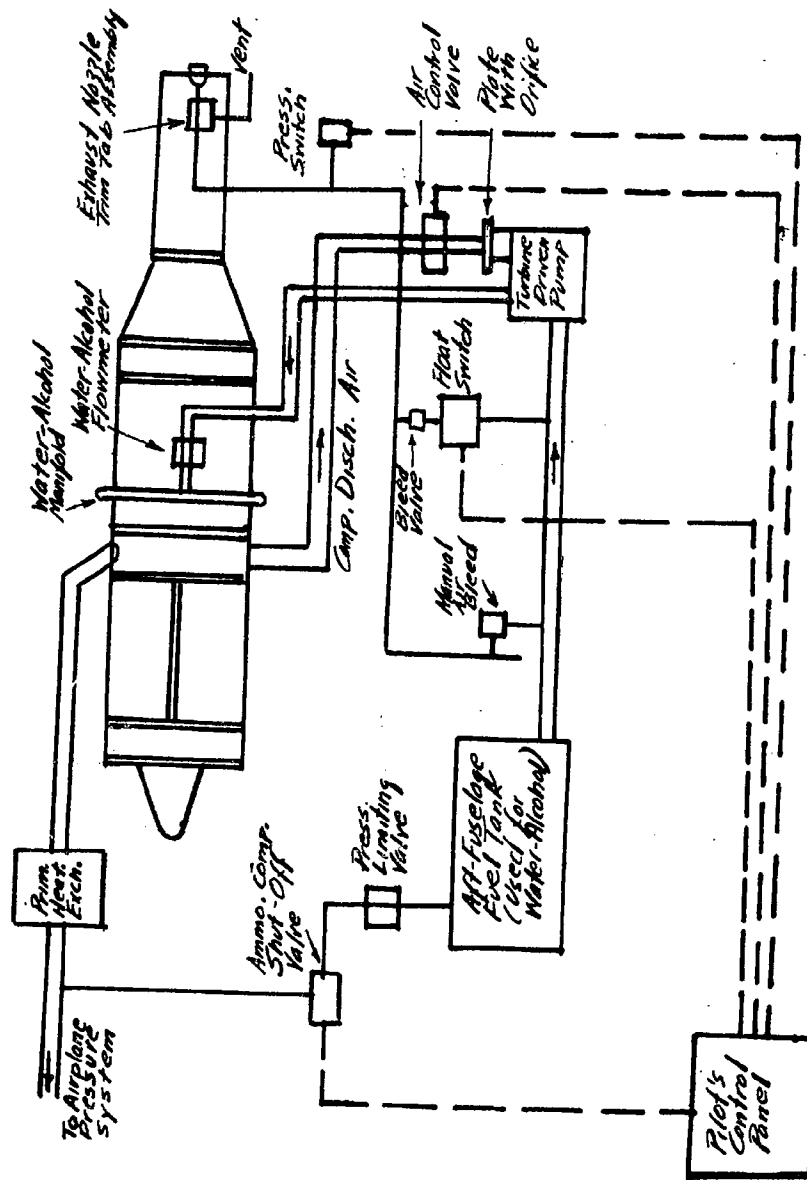


FIGURE 15. Schematic Diagram Of The J47-GE-27 Engine Water-Alcohol Injection System Installation In The F-86F Aircraft.

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mately half the test the standard 100 gal/hr water-alcohol nozzles were used, and for the remainder of the test the experimental 50 gal/hr nozzles were used with the lower capacity pump.

The system was modified only slightly for the F-86F aircraft test installation. In that aircraft, the ammunition compartment heating system was used to supply tank pressurization. The connection to the system was made just downstream from the heating air shut-off valve and wiring was installed so the valve could be controlled from the cockpit. A larger capacity turbine pump was installed to overcome the difficulties encountered in the F-86A aircraft installation. A plague of pump failures began, apparently caused by cavitation which resulted in overspeeding. Various modifications were made in a vain effort to eliminate air from the water-alcohol pump inlet and outlet lines. Eventually the water-alcohol supply line between the tank and the pump was changed completely. A plate was made to fit in place of the tank inspection door on the aft face of the tank. A tube, welded to the plate, extended in to the center of the tank, and a mating line was connected to the pump inlet. By keeping the line as low as possible, trapped air was held to a minimum. In addition, a valve was installed so that all air could be bled from the line after each servicing; a slightly smaller capacity pump was also installed to replace the large capacity pump.

The J47-GE-27 engine in the F-86F aircraft was not normally equipped for water-alcohol injection. The J47-GE-27 engine used for the testing was therefore modified by installing an external water-alcohol manifold, eight flex lines for connecting the manifold to each combustion chamber, and a set of combustion chambers from a J47-GE-25 engine which was used in a B-47 type aircraft. The water-alcohol manifold encircled the forward end of the combustion system and was attached to the compressor rear frame. The experimental 75 gal/hr water-alcohol nozzles were used throughout the test.

The pilot's water-alcohol injection control panel consisted of two switches for opening and closing the tank pressurization shut-off valve and the turbine pump air control valve, appropriate circuit breakers, and a light which indicated when the pressure switch closed; Figure 16 il-

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illustrates the water-alcohol injection control presentation located in the cockpit. To initiate water-alcohol injection, the tank was first pressurized. After at least 15 seconds the water-alcohol injection control switch was moved to PRIME and held, energizing the circuit to the air control valve providing the float switch was closed. The float switch was a safety feature installed to prevent starting the pump until there was a head of water-alcohol at the pump inlet. A air bleed valve was used in conjunction with the float valve to allow air to escape. Pressurization of the water tank forced water into the float valve assembly, raising the float and closing the switch and at the same time closing the bleed valve. When the motor on the air control valve was energized, the valve opened allowing compressor discharge air to energize the turbine pump. As soon as pump discharge pressure was 10 psi greater than combustion chamber pressure, water-alcohol was forced through the check valve and injection was started. Simultaneously the exhaust nozzle tab, which was necessary to maintain temperature, was forced up into the exhaust stream by the force of the pump discharge pressure on the piston in the nozzle actuator. Figure 17 illustrates the tab assembly used for the testing. As soon as the water-alcohol pressure reached the pre-selected pressure switch setting it closed the switch energizing the pilot's indicator light and completed an alternate circuit to the air valve. The pilot then released the switch and water-alcohol injection continued until the pump discharge pressure dropped to the level at which the pressure switch was set to open. Opening of the pressure switch de-energized the air valve and stopped the airflow to the pump. If the pilot desired, he could stop water-alcohol injection by moving his control switch to the OVERRIDE OFF position; otherwise, the injection continued until the water-alcohol mixture was expended.

In 35 flight hours on the test F-86A aircraft, approximately three (3) hours of water-alcohol injection time was accumulated. For the test F-86F aircraft, in nearly 50 hours of flight testing, over three (3) hours of water-alcohol injection was accomplished. Many more hours of water-alcohol injection were accumulated during ground tests and some initial flight testing utilizing a B-45 aircraft was accomplished.

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C. Results

In general, the first water-alcohol flight tests utilizing a single J47 engine of a B-45 aircraft were successful and most promising. A difficult task came however with the adapting of the system to the F-86 aircraft.

The augmentation obtained with the configuration tested in the F-86A aircraft utilizing a J47-GE-13 engine was very encouraging, being approximately 30% at altitudes of 30,000 feet and above. No serious problems were encountered, and operation was for the most part quite satisfactory at altitudes up to 40,000 feet. At higher altitudes the termination of water-alcohol injection during climbs invariably resulted in flame-outs, although level flight operation was normal. Upon disassembly of a test J47-GE-13 engine for inspection after approximately 1-1/2 hours of water-alcohol injection time, the combustion chamber inner liners and transition liners were found to be damaged. The damage was attributed to the poor and erratic spray pattern exhibited by the standard water-alcohol nozzles in the J47-GE-13 engine. During an equivalent period of operation with lower flow capacity nozzles, no engine damage was incurred; however, the restriction of these nozzles was so great that flow, and consequently augmentation, was noticeably reduced even with maximum power input to the pump. Since a method of augmentation for the F-86F aircraft was of primary interest, and because more intensive testing was accomplished with that aircraft, further discussion will be confined to that phase of the testing.

Figures 18 and 19 show comparative climb performance data between a standard F-86F aircraft and one utilizing water-alcohol injection. It can be observed from Figure 18 that the rate-of-climb of a water-alcohol augmented F-86F is continually increased above an altitude of 25,000 feet until it is over double the dry rate-of-climb at 40,000 feet. As a consequence, the time to climb from an altitude of 20,000 feet to an altitude of 30,000 feet is reduced by approximately one minute. Also, the time to climb from an altitude of 30,000 feet to an altitude of 40,000 feet is reduced by over four minutes. The reason for the discontinuities in Figure 19 is that a change in water-alcohol flow rate had to be made for each range of

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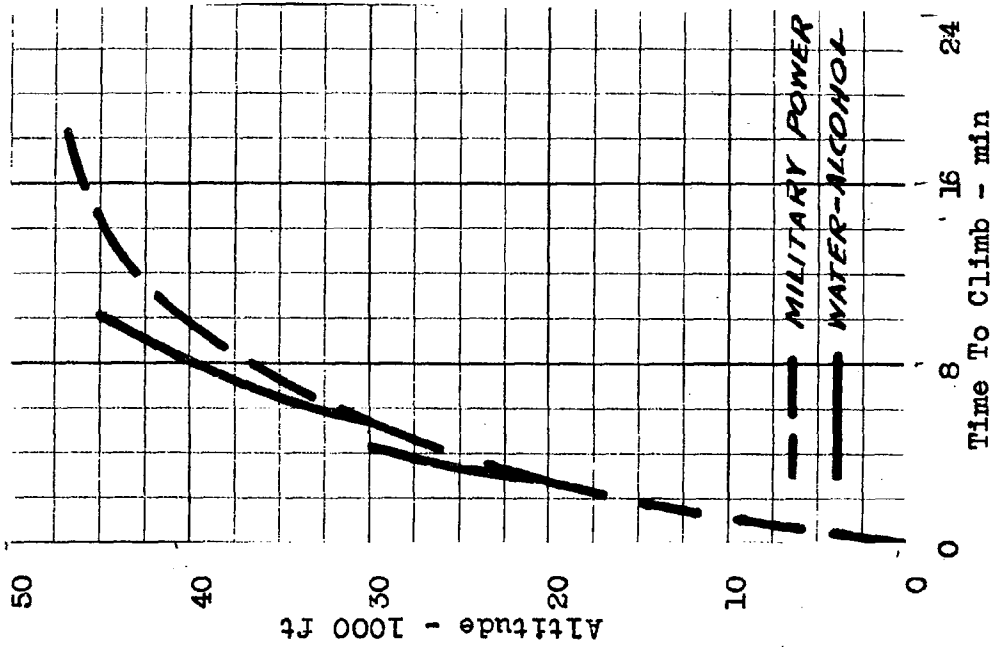


FIGURE 19. F-86F Aircraft. Combat Weight. Clean.

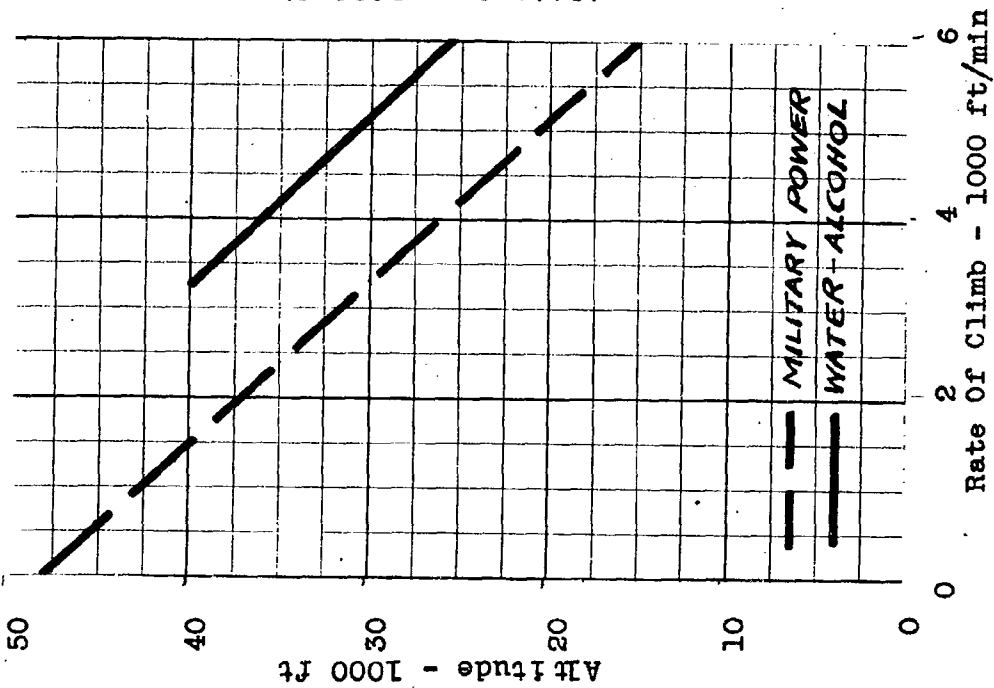


FIGURE 18. F-86F Aircraft, Combat Weight. Clean.

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altitudes; such a change could only be made on the ground. Figure 20 shows the increase in level flight true air speed that was obtained by the use of water-alcohol injection; it can be seen that the speed of the F-86F was increased by 10 knots at an altitude of 20,000 feet and nearly 15 knots at an altitude of 45,000 feet. Maximum thrust augmentation obtained in the F-86F aircraft varied from approximately 19% at 20,000 feet to approximately 29% at an altitude of 40,000 feet; such data are presented in Figure 21. Since changing the water-alcohol injection rate in order to extend operation to higher altitudes was a ground adjustment, the reduction in flow necessary at high altitude resulted in less flow and less augmentation at the lower altitudes if the system was set for high altitude operation. The augmentation at the lower altitudes was reduced in the order of 20% so as to allow satisfactory operation at the higher altitudes and eliminate the need for an adjustment. It was reasoned that satisfactory operation at altitudes above 40,000 feet was worth the loss in augmentation which resulted at the lower altitudes. Figure 22 shows comparative data gathered from accelerations from minimum level flight true air speed to maximum level flight true air speed at an altitude of 35,000 feet. It can be noted that with water-alcohol injection, the F-86F aircraft reaches the same maximum true air speed at an altitude of 35,000 feet that was possible with an unaugmented F-86F aircraft approximately 1-1/4 minutes sooner.

More serious engine instability was encountered with the J47-GE-27 engine than with the J47-GE-13 engine, and alcohol percentages were more critical. Mixtures ranging from 20% to 28% alcohol were used, but with the higher percentages there was a tendency for the engine to overspeed when starting or stopping water-alcohol injection. The maximum water-alcohol flow schedule used, as shown in Figure 23, which varied from 48 gal/min at an altitude of 20,000 feet to 33 gal/min at an altitude of 40,000 feet gave optimum performance throughout that altitude range with a 24% alcohol mixture. Above an altitude of 40,000 feet, that combination resulted in flame-outs when the injection was terminated under conditions other than level stabilized flight. By reducing the flow rate to 26 gal/min at an altitude of 40,000 feet, satisfactory operation was obtained during level flight, climbs, dives, and other

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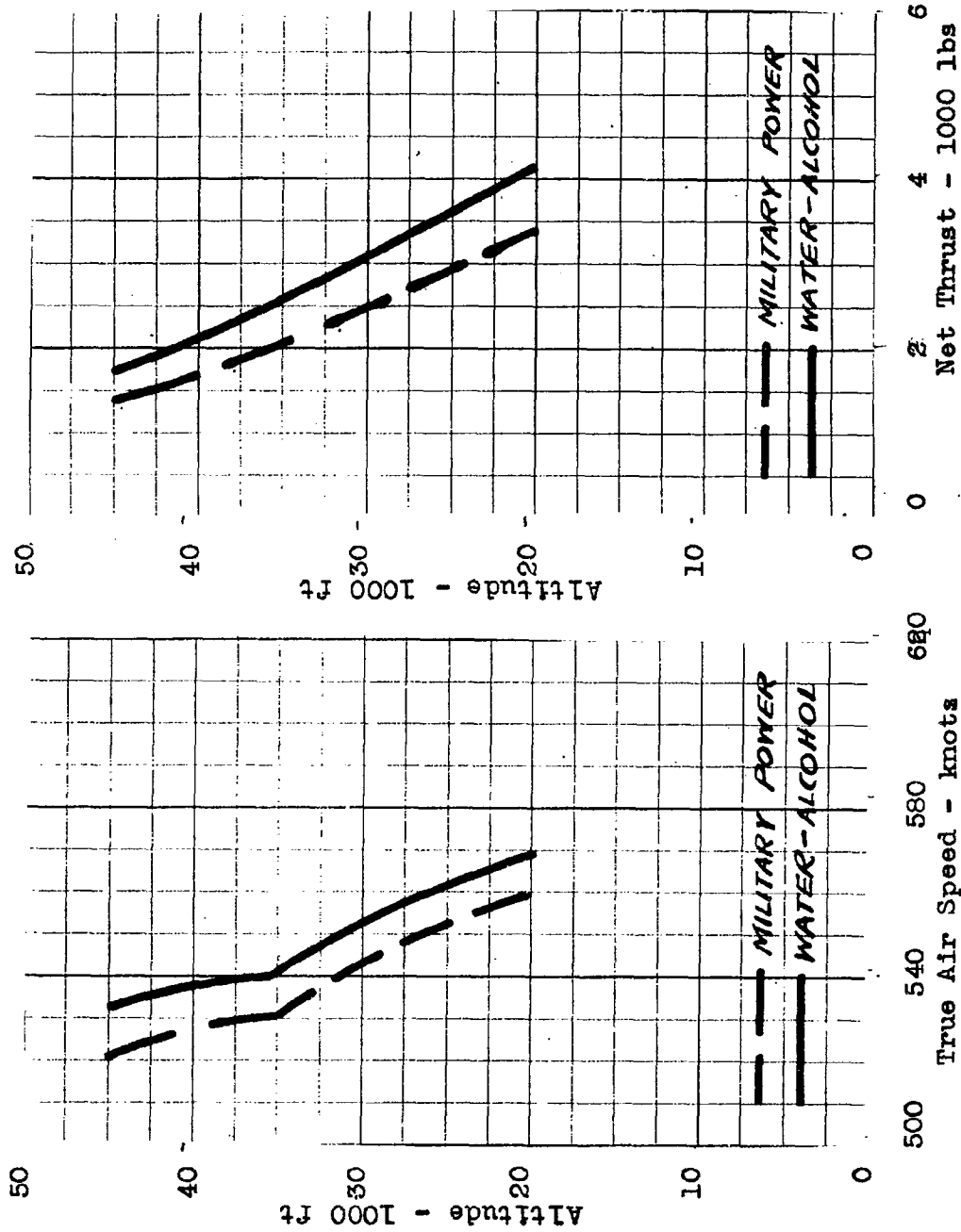


FIGURE 21. J47-GE-27 Engine.
F-86F Installation.

FIGURE 20. F-86F Aircraft. Combat Weight. Clean.

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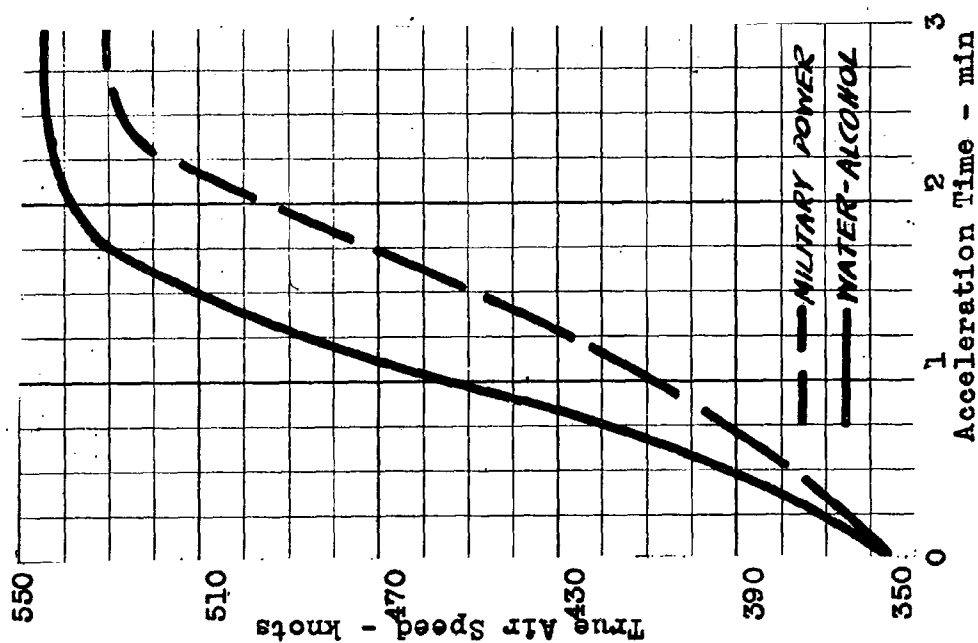
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FIGURE 22. F-86F Aircraft. Combat Weight. Clean. Altitude 35000 ft.

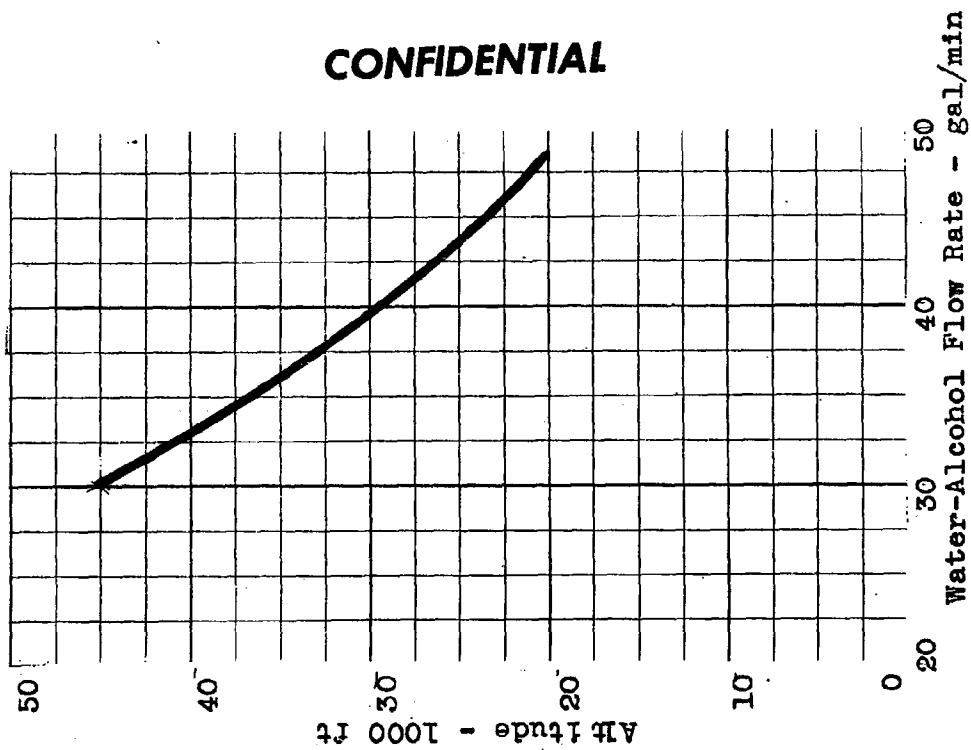


FIGURE 23. J47-GE-27 Engine. F-86F Installation.

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maneuvers at altitudes up to 48,000 feet; only one climb was continued to an altitude of 50,000 feet. Operation above 45,000 feet necessitated that the pilot more closely monitor the exhaust gas temperature so as to keep it within limits. Also at altitudes of 45,000 feet and above there was usually surging in engine speed when water-alcohol injection was initiated but throttle adjustment could alleviate the problem. The single exhaust nozzle tab, as located, caused an objectionable yaw force, but it was thought that by a redesign or the symmetrically locating of twin tabs would eliminate the trouble.

It was concluded that the normal combustion system components of the J47-GE-27 engine have a water-alcohol injection endurance life of approximately two hours. It is noted from the forgoing discussion that water-alcohol augmentation of the J47-GE-27 engine in the F-86F aircraft offered substantial gains in performance. With the exception of the 105 gallon fuel tank converted to carry water-alcohol, the configuration tested was satisfactory; a single aircraft under-fuselage tank was designed, although never tested, to overcome that difficulty. It was apparent however that some additional testing was necessary. It was also apparent that even with ultimate refinement, the gross weight of the aircraft would be considerably increased. One problem that appeared difficult in light of the fact that operation was to be in Korea, was that of logistics. It was decided that a method of augmentation that did not require another fluid and therefore not necessitate a dual tank system would be the most desired, thereby, allowing flexibility from mission to mission or as the need occurred during any one mission. Since it was not possible to support a high-priority effort on more than one project, it was concluded that pre-turbine injection, which looked promising on the basis of preliminary analysis, would be concentrated on and that further work on water-alcohol injection would be continued on a development basis with direct application being aimed at the B-47 aircraft for take-off only, where the control problems would necessarily be less. It was felt that a water-fuel injection system might be used. With such a system JP-4 would be substituted for the alcohol.

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SECTION V

PRE-TURBINE INJECTION

A. General

One of the most popular means of augmenting the thrust of a turbojet engine is afterburning. An afterburner for the J47 engine had already undergone some development testing as early as February 1948; this afterburner development engine was designated the XJ47-GE-5. Further development of that engine led to the J47-GE-17 engine which powers the F-86D aircraft and eventually to the J47-GE-33 engine. The J47-GE-17 and the J47-GE-27 engine were by no means interchangeable. Even provided it would have been possible to install a conventional afterburner in the F-86F aircraft, the added pressure losses when non-afterburning would have reduced the aircraft's cruising range. Since the cruise-out portion of the missions in Korea were lengthy, the use of a conventional afterburner would show a disadvantage from that viewpoint.

A method of afterburning was necessary which would, in addition to providing the increase in thrust necessary, also be capable of being incorporated in the existing aircraft with little modification and also be such that the dry engine performance of the aircraft would not be affected. With such rigid requirements, only pre-turbine injection seemed feasible. Pre-turbine injection (hereafter referred to as PTI) is a system of reheat whereby the fuel for afterburning is injected upstream of the turbine in contrast to the conventional method of injecting the fuel for afterburning downstream of the turbine. With such a system the turbine wheel acts as a flameholder rather than having separate flameholders which contribute to the dry loss of an afterburning engine. PTI, which in a liberal sense might be referred to as an extremely short afterburner, allowed the combining of the diffuser and burner sections into one and therefore made installation in the already existing F-86 aircraft possible. The first work on such a system was conducted in Germany as early as October 1939. The first successful tests were run on the German Jumo 004 turbojet engine in 1942. Later, although prior to the out-

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break of hostilities in Korea, a series of similar tests were conducted on American engines of a later design in the Power Plant Laboratory, Wright Field.

Many of the problems which had to be resolved during PTI development and testing were similar to problems which were encountered in previous afterburner developments. These problems however were aggravated by the limitations placed on the system in the form of weight and available space restrictions. Since PTI was to be installed in the F-86F aircraft, a rather large portion of the testing had to be centered around obtaining adequate cooling of the aircraft structure. Two test aircraft were used, one was devoted to engine testing and the other toward resolving the problems associated with the installations.

B. Testing

Thrust augmentation during PTI operation was obtained by a combination of afterburning and basic engine overtemperature operation. Naturally basic engine overtemperature was undesirable but it was nevertheless necessary because of the space limitations in the aft fuselage of the F-86 aircraft. Had space been available, a larger tail-pipe and nozzle could have been utilized, much like a conventional afterburner, and the increase in back pressure on the engine due to afterburning could have been reduced. It is noted that afterburning has much the same effect as closing the exhaust nozzle of an engine. A decrease in the exhaust nozzle area increases the back pressure on the turbine which reflects forward through the engine to the compressor outlet, resulting in an increase in compressor pressure ratio. In order to maintain the additional work from the turbine that is required to maintain engine operation at this increased compressor pressure ratio condition, the turbine inlet temperature must be increased. Such was the case with PTI where it was not possible to increase the exhaust nozzle area as much as was desired and still maintain stable burning. Without overtemperature, the engine rpm would continually drop off with increasing altitude. The degree of engine overtemperature for which the PTI system was designed was based upon an estimated five hour turbine bucket life. However, in order to obtain stable PTI burning, it was necessary to operate at augmentation ratios higher than those originally intended with a conse-

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quent adverse effect on engine life. A variable-area exhaust nozzle was used to maintain the correct combination of pressure and temperature for stabilized PTI burning in the engine exhaust section as well as to provide the correct nozzle area for dry engine operation. In order to minimize control system design complexities and to maintain the fuel flow requirements within the capacity of the existing fuel pumps, PTI operation was limited to altitudes above 20,000 feet. Such requirements which would allow operation below an altitude of 20,000 feet were beyond the limits of the equipment used in the PTI system.

The PTI system is shown schematically in Figure 24. Fuel for pre-turbine injection was supplied from the normal aircraft fuel system by the standard engine emergency fuel pump. A PTI metering valve regulated the fuel flow in accordance with the pressure schedule set by the EC-2 emergency fuel regulator normally provided on the engine. The metered PTI fuel was injected into the gas stream by means of four probes located in alternate combustion chamber transition liners forward of the turbine nozzle. The fuel vaporized and burned downstream of the turbine wheel. Operation was such that flameholders were not thought necessary. Naturally when the correct combination of pressure, temperature, and fuel-air ratio occurred, burning would take place. The fuel in order to burn had to enter a zone that would continually meet the combustion requirements. Thus the conditions at turbine outlet had to meet such requirements. Should combustion not have taken place, the PTI fuel would have merely passed out through the engine unburned. Once lit, the flame did not progress upstream as the gas velocity far exceeded the rate of flame propagation. At first the system utilized a hot streak ignition system but subsequent flight tests demonstrated that it was unnecessary for satisfactory light offs. Initial flight testing was accomplished using a water injection type tab to vary the exhaust nozzle area pending development of a suitable variable-area nozzle and nozzle control. Early tests were run with the standard F-86F tail-pipe and it was not adequate due to the high failure rate. Numerous tests, with tail-pipes constructed of various materials and thicknesses, were conducted. Since failures were caused by excessive temperature of the tail-pipe skin, some method of lowering the skin temperature was sought. The initial testing was conducted with an insulation blanket surrounding the tail-pipe. Radiation heat shields were added to several major

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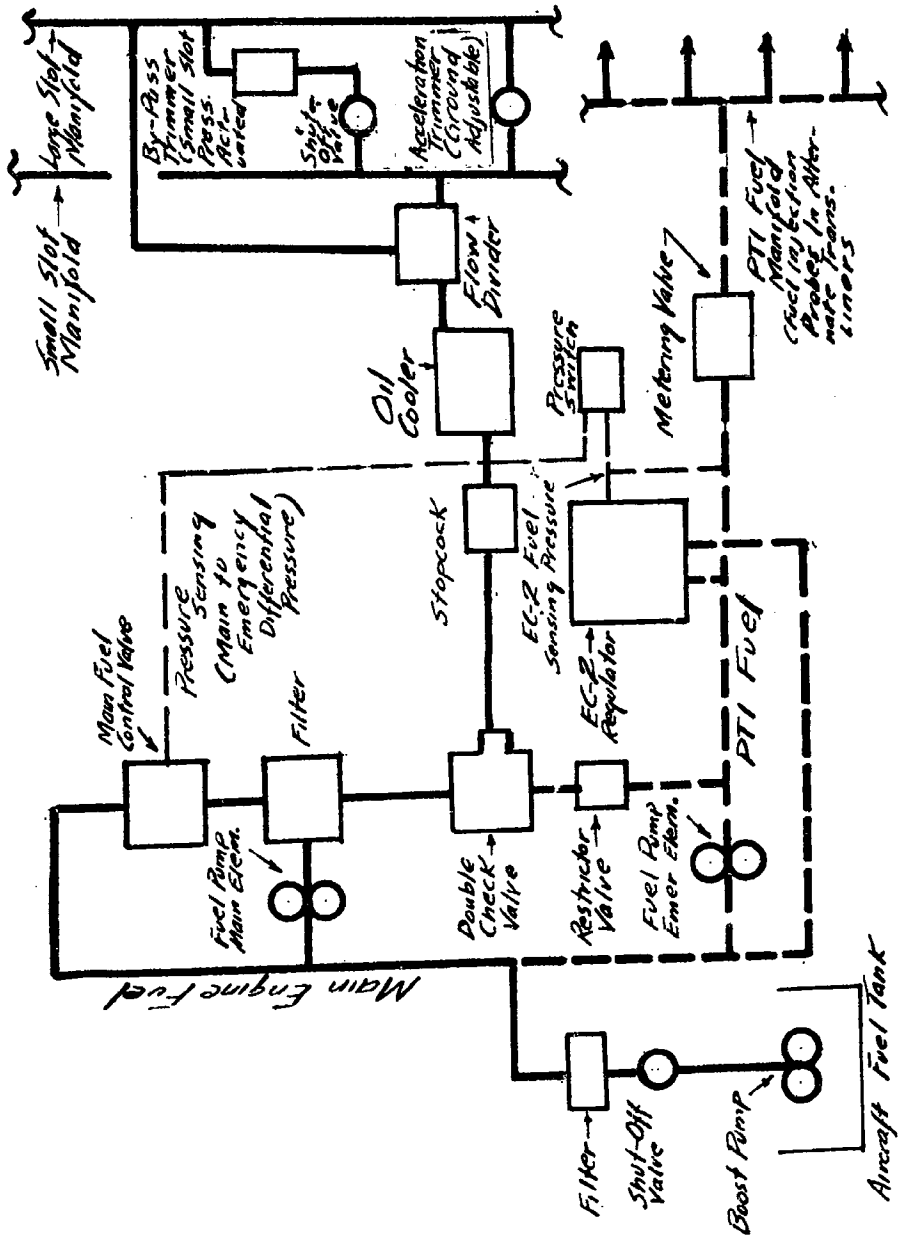


FIGURE 24. Schematic Diagram Of The J47-GE-27 Pre-Turbine Fuel Injection System Installation In The F-86F Aircraft.

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aircraft frames and local cooling was provided for the aft canted frame. Ram air scoops were incorporated to increase the cooling airflow through the aft fuselage. High tail-pipe skin temperatures measured during flights indicated that a blanketed tail-pipe configuration was unsuitable and that the temperature could be lowered by about 150° to 200°F by their removal. The tailpipe blankets were thus replaced by a silver plated shroud. The cooling problem was made difficult by the partial loss of ejector action due to the decrease in secondary air area as a result of installing a variable-area nozzle which was necessarily larger than the standard tail-pipe's nozzle. Further reduction in the temperatures to which the tail-pipe was exposed became possible with the introduction of a corrugated louvered liner. The liner was installed for better cooling and not to prevent screech since that problem was not present. The liner was identical in construction, although not in size, to the liner in the afterburner of the J47D series engines used in the F-86D aircraft. Tail-pipe construction was revised to include a flex joint between the engine tail-cone and tail-pipe to eliminate the high overhang moment of this tail-pipe configuration.

A by-pass fuel line with a variable trimmer valve was specially provided between the engine small and large slot manifolds to increase the basic engine fuel flow during PTI operation. A solenoid shut-off opened the by-pass line when PTI was selected. The variable trimmer valve is a metering device which varies the by-pass flow in accordance with small slot fuel pressure, closing completely above approximately 45,000 feet altitude where the main engine regulator fuel schedule is adequate to supply engine fuel demands. A second special by-pass fuel line with a ground adjustable trimmer (needle valve) was provided between the small and large slot manifolds for ease in adjusting the engine acceleration schedule on the ground.

After a review of many variable-area nozzle configurations, the flat plate orifice type appeared to best suit all needs since it was simple and relatively light in weight. Actuation was obtained at a single circumferential point thus reducing the space requirements. Actuation loads were relatively light since only friction had to be overcome. The power source for the nozzle was a 1/6 horsepower air-

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frame mounted electric motor. The motor actuated a ball bearing jack screw mounted on the nozzle housing. Figure 25 is a sketch of the nozzle arrangement. It was found that one nozzle position would suffice for PTI operation but two were needed for dry operation; this was due to the fact that the flow coefficient and effective areas varied appreciably at the low actual values. There were two possible ways of compensating for the variations of these factors -- constantly variable or a step control nozzle. The simplest method and also the most advantageous from a development time standpoint was step control. Ideally, for the most effectively controlled engine, a constantly variable-area nozzle was desirable since any use of a step control would necessarily compromise performance. It was decided that only a fully variable-area nozzle would be satisfactory, for with a step control a three position nozzle was necessary; one for take-off, one for climb and normal performance at altitude, and one for PTI operation.

The fully variable-area nozzle used in conjunction with PTI was controlled by a pressure sensing device known as the Micro-Jet. The unit controlled turbine discharge temperature by varying the nozzle position to maintain a constant pressure ratio across the turbine for any given operating condition. The Micro-Jet contained a diaphragm, one side of which was exposed to turbine discharge pressure through a sensing line. The other side of the diaphragm was exposed to a pressure which was controlled by bleeding compressor discharge air through a fixed inlet orifice and a variable discharge orifice. The fixed orifice was analogous to the engine's turbine and the orifice size was ground adjustable to give the proper turbine pressure ratio (determined by turbine discharge temperature). The second orifice simulated the variable-area exhaust nozzle. A tapered needle in the variable orifice was automatically positioned by the diaphragm to relieve any pressure differential across the diaphragm by increasing or decreasing the pressure drop through the orifice. Should the needle be disturbed from its neutral position (determined by the pre-set turbine pressure ratio) by a pressure differential across the diaphragm, the electrical contacts of the Micro-Jet would close signalling an electrical control box to energize the nozzle actuator motor to either close or open the exhaust nozzle (depending on the direction of needle movement). The nozzle closed or opened

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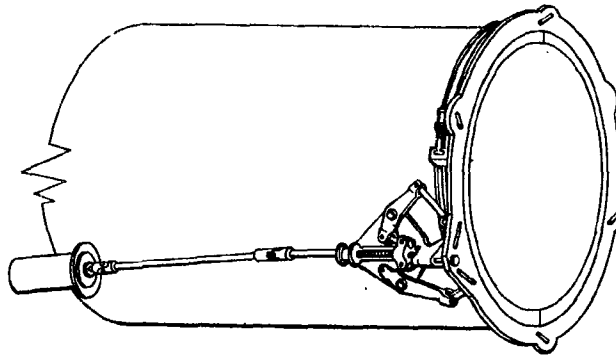


FIGURE 25. F-86F Tail-Pipe Variable Area Nozzle Utilized In The J47-GE-27 Engine Pre-Turbine Fuel Injection Tests.

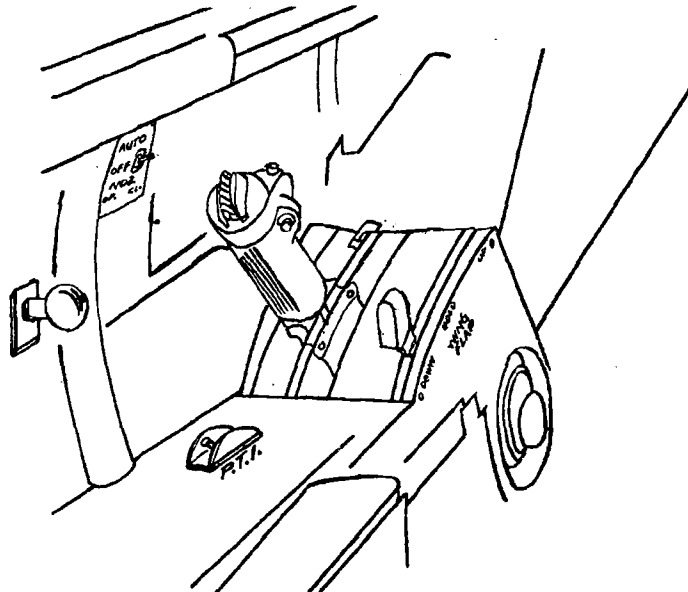


FIGURE 26. J47-GE-27 Engine Pre-Turbine Fuel Injection Control Cockpit Presentation In The F-86F Aircraft.

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until it reached a position which restored the turbine pressure ratio to the value determined by the neutral position of the needle.

A solenoid operated bleed valve on the compressor discharge pressure side of the Micro-Jet diaphragm, which was open during dry operation and closed during PTI operation, permitted engine operation on two different turbine pressure ratios thus allowing the control to be used for both dry and PTI operation. Since retarding of the throttle below its military power lowered the turbine pressure ratio, the control tended to open the nozzle as the throttle setting was reduced. The light-off problem at altitude was considerably alleviated with the introduction of the Micro-Jet control.

A PTI pressure cut-out switch was provided as a means of de-energizing the PTI circuit to prevent engine overspeed should override of the main fuel control system by the emergency fuel control system have occurred during PTI operation. Figure 26 is a sketch of the PTI presentation in the cockpit. PTI operation was initiated by the pilot first setting the guarded PTI ready switch to ON, he then set the nozzle selector switch to AUTOMATIC and then followed by moving the throttle to the military power position and then momentarily outboard from the military power detent. PTI light-up and operation was fully automatic following the above three step procedure. However, the pilot had to observe the PTI exhaust gas temperature limit of 1200°C during PTI operation. If the temperature approached the limit, the throttle had to be retarded. PTI would remain in operation until the throttle was retarded to the 96% rpm position, but below 96% rpm PTI was automatically shut off. If for any reason the pilot wished to stop PTI operation, either the PTI ready switch could be positioned to OFF or the throttle retarded past the 96% rpm position.

During 894 hours of static testing, 117 hours of PTI were accomplished. In addition 46 hours of testing were accomplished in an altitude tank with 8 hours of PTI being accomplished. A total of 209 flights were conducted utilizing two aircraft; during the 135 hours of flight testing, 17 hours of PTI were accomplished. A single 50 hour endur-

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ance test and three consecutive 50 hour tests were run. During the single 50 hour endurance test 100 minutes of PTI were accomplished. Five hours of PTI were accomplished during the three consecutive 50 hour tests. The maximum altitude reached during the testing was 53,760 feet. Twelve flights were conducted above an altitude of 50,000 feet and 26 flights were conducted above an altitude of 45,000 feet.

C. Results

PTI applied to the J47-GE-27 engine in the F-86F aircraft considerably improved the weapon's performance. Pertinent data concerning PTI are shown in Figures 27 through 34. Also shown are performance data for a production F-86F aircraft for purposes of comparison.

As can be seen from Figure 27, the rate-of-climb of the F-86F aircraft using PTI, as compared to a standard unaugmented F-86F, was more than doubled at an altitude of 35,000 feet, tripled at an altitude of 40,000 feet and was increased by as much as four times at an altitude of 45,000 feet. Thus the time to climb from an altitude of 20,000 feet to an altitude of 45,000 feet was reduced by over six minutes, the comparative times to climb being approximately 4 and 10 minutes for a PTI equipped aircraft and a production aircraft respectfully. A PTI climb could be made from an altitude of 20,000 feet to an altitude of 50,000 feet in approximately 6 minutes; Figure 28 presents such data. The maximum level flight true air speed of the F-86F aircraft was increased in the order of 20 to 25 knots between altitudes of 20,000 and 45,000 feet; Figure 29 shows this increase. The thrust augmentation obtained by using PTI, as can be seen from Figure 30, showed as average increase of about 45% over the 20,000 to 45,000 feet altitude range. As a result of PTI, a substantial improvement in acceleration was made possible as evidenced from Figure 31. Thus with PTI, the maximum level flight true air speed of a production F-86F aircraft at an altitude of 35,000 feet was reached over one minute sooner. Figure 32 is a compilation of specific fuel consumption data; the Figure shows the rather large increase in specific fuel consumption with altitude when using PTI. As can be observed, the unaugmented J47-GE-27 engine's specific fuel consumption is nearly constant over the complete altitude range while the spe-

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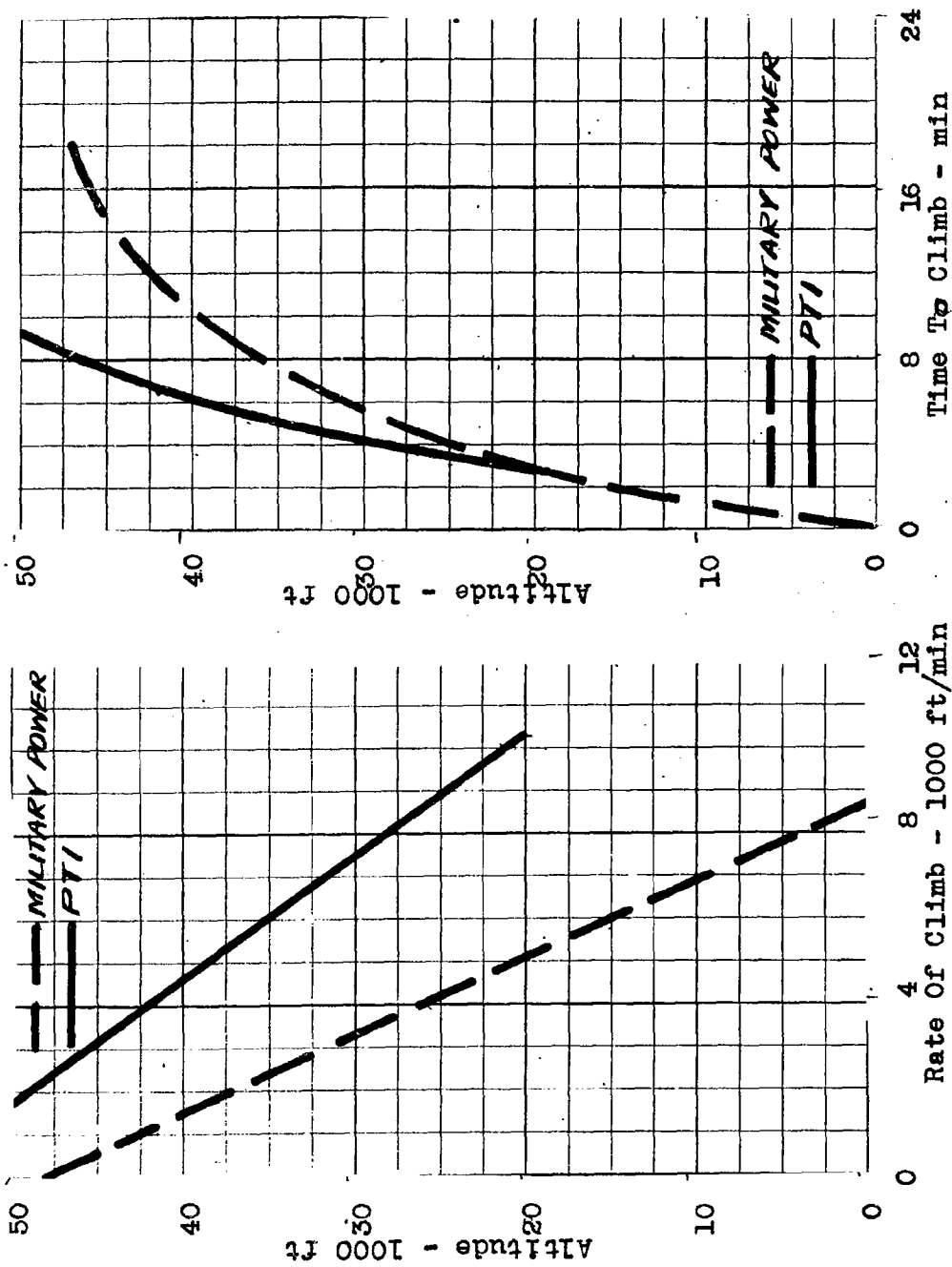
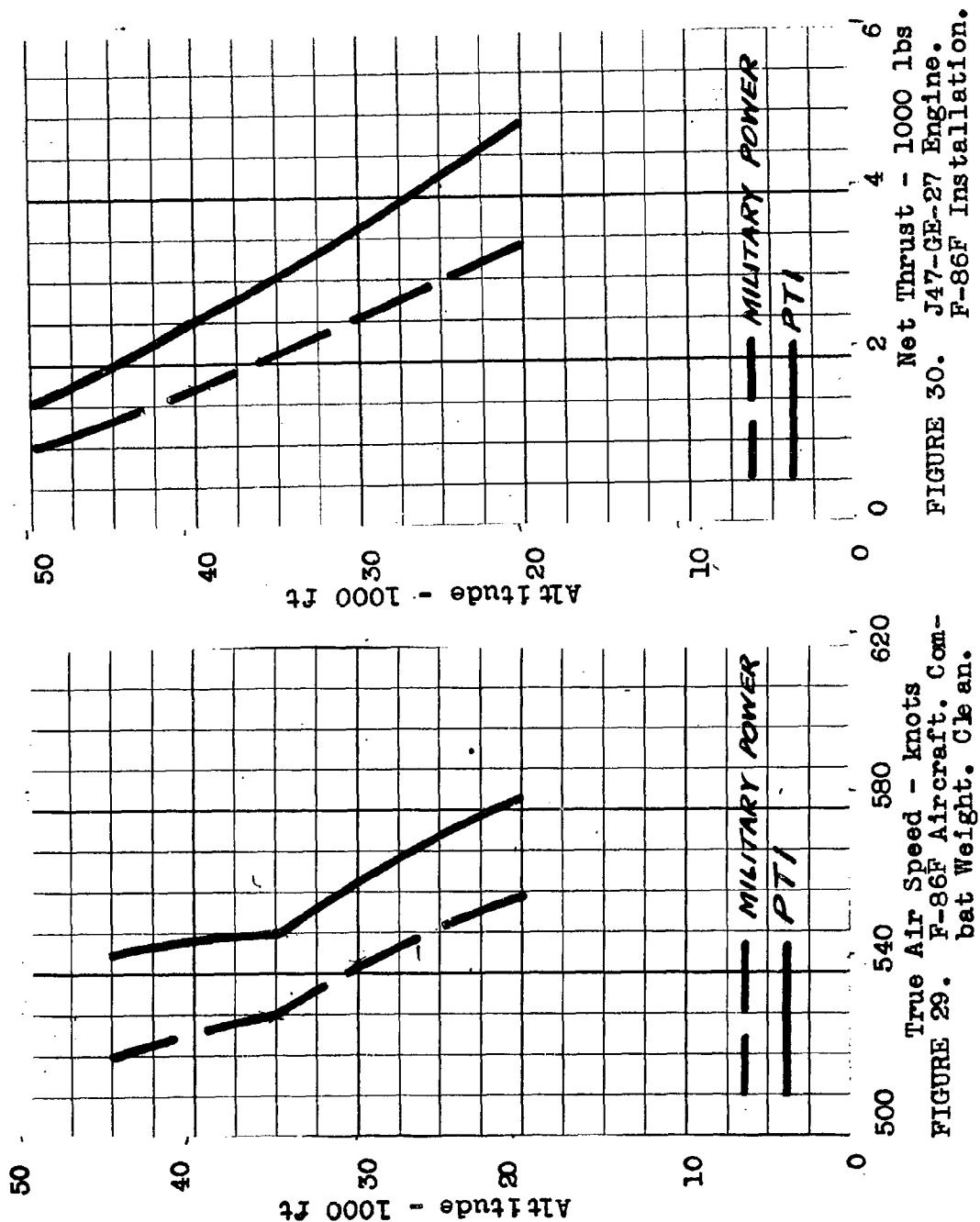


FIGURE 28. F-86F Aircraft. Combat Weight. Clean.

FIGURE 27. F-86F Aircraft. Combat Weight. Clean.

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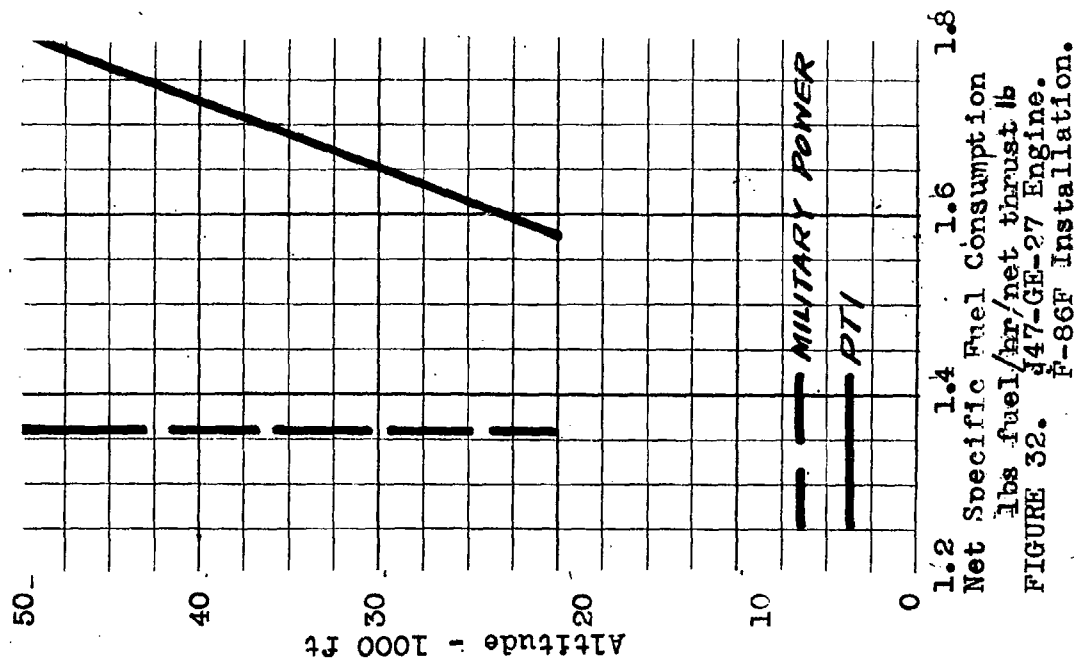


FIGURE 32. F-86F Installation.

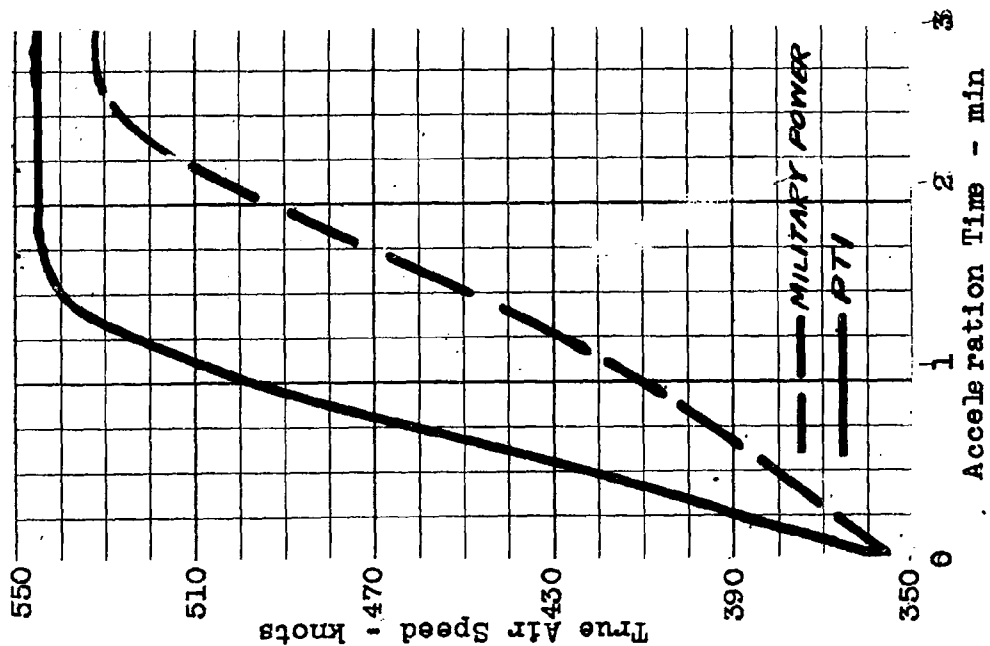


FIGURE 31. F-86F Aircraft. Combat Weight. Clean Altitude 35000 ft.

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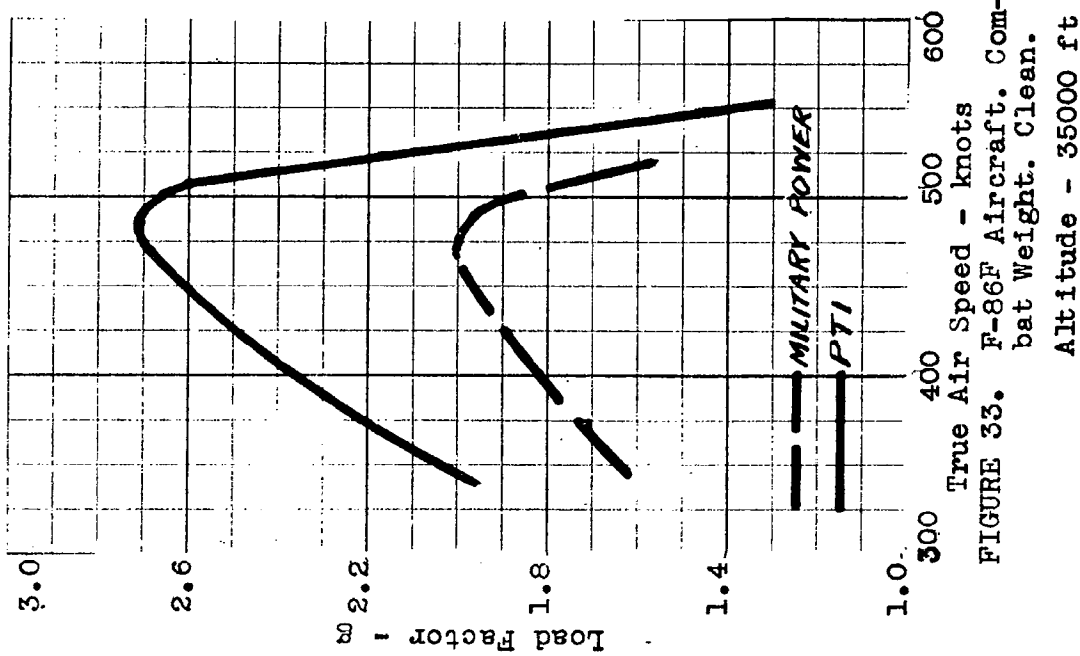
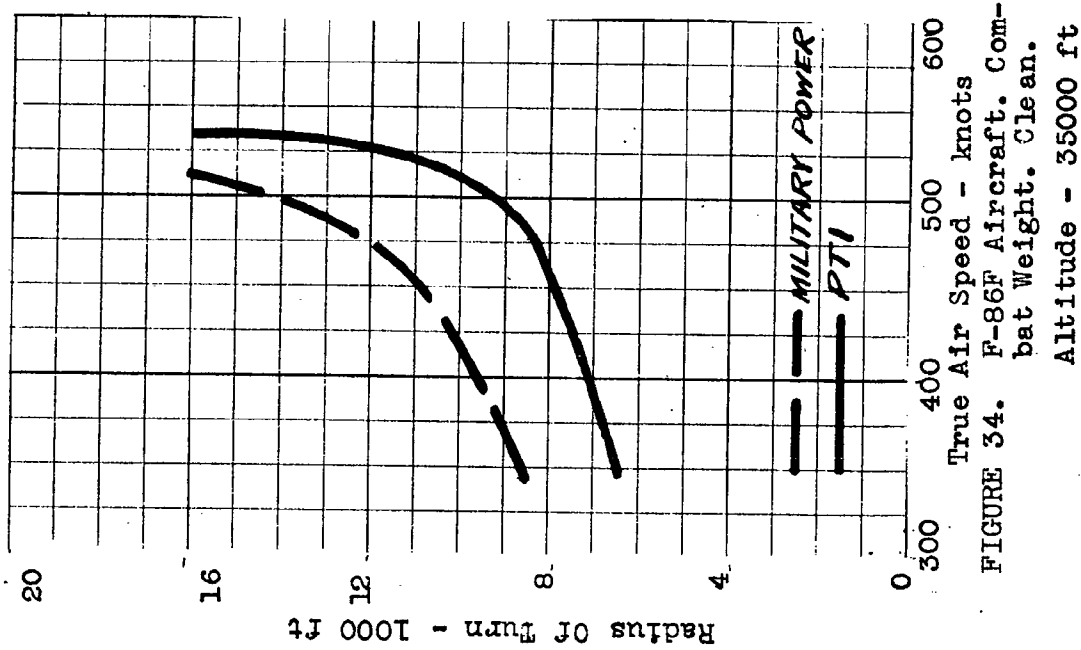
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cific fuel consumption, utilizing PTI, increases due to the necessity of having to increase quite substantially the fuel flow at the higher altitudes. Nevertheless, the specific fuel consumptions obtained with PTI were considerably better than those obtained with a standard afterburning engine; for instance, the J47-GE-17 engine which powers the F-86D aircraft has a specific fuel consumption approximately 40% greater than a PTI equipped J47-GE-27 engine. It was concluded that the combat fuel requirements of PTI would not impose a serious operational limitation; the combat radius of the F-86F aircraft utilizing PTI was reduced even less due to the addition of the increased weight of the system. The J47-GE-17 engine utilizing a standard afterburner gives a higher augmentation ratio than the PTI equipped J47-GE-27, but also at a cost of nearly five times the weight. With PTI, there was an increase in aircraft weight of 140 pounds. The PTI kit itself actually weighed 210 pounds; however 70 pounds of existing parts were deleted. For instance, the tail-pipe and nozzle included in the kit replaced those already in the aircraft. The afterburner of the J47-GE-17 engine alone weighs 655 pounds.

A pronounced improvement in airplane maneuverability, as can be observed from Figures 33 and 34 was possible with the additional thrust provided by PTI operation. Altitude turns and maneuvers with PTI could be performed with less drop-off in speed and altitude. Also, constant altitude, constant speed maneuvers could be accomplished with higher load factors and reduced turning radius with PTI. One disturbing factor arose however due to the added weight of the PTI installation. The addition of the PTI installation plus ballast, combined with the expenditure of ammunition and fuel sequencing produced unacceptable loading conditions. With the PTI system tested, 150 pounds of ballast were required to provide the same acceptable aircraft balance as an unmodified aircraft. The added weight of the PTI installation caused a rearward shift of the aircraft's center of gravity such that it exceeded the aft neutral stability limits when the ammunition was expended and thus the ballast in the nose of the aircraft was necessary to correct that condition; but, when the ballast was added and the ammunition was retained, the center of gravity shifted forward to its maximum in-flight position. The net result being an approximate 15% reduc-

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tion in the aircraft's maximum allowable load factor. It was felt that with the addition of the proposed 12 inch wing tip extension, the 150 pound ballast could be deleted. The 12 inch wing tip extension would allow an approximate 3% improvement in aircraft stability.

The degree of engine overtemperature for which the PTI system was designed was based upon an estimated five hour turbine bucket life. However, in order to obtain stable PTI burning at the higher altitudes, it was necessary to operate at augmentation ratios higher than those originally intended with a consequent adverse effect on engine life. Nearly 40% of the augmentation obtained was due to overtemperature of the basic engine itself. Turbine buckets during the test program had to be replaced nearly every one hour of PTI operation. The buckets were replaced because of excessive growth or actual failure. It was necessary to replace the turbine wheel after approximately three hours of PTI operation. Throughout the entire test program, five turbine wheel failures were experienced and several others were rejected after inspection. The exhaust-cone and tail-pipe had to be replaced on the average of about once every tenth flight. Some difficulty was experienced with the variable-area nozzle as it had a tendency to stick; the difficulty was overcome by hardening of the segments. Burner roughness was found at altitudes near 50,000 feet, but in general the entire system functioned satisfactorily and burning and light-ups were smooth. One very great problem centered around the obtaining of the proper fuel scheduling and many flights were accomplished to obtain satisfactory operation, particularly at the higher altitudes. Since the EC-2 standard emergency fuel regulator was used as the PTI fuel metering device, maintaining a close tolerance on the fuel injection rate was impossible and that complicated with the narrow operating range of the burner at high altitudes made the problem more difficult. Although the emergency fuel regulator was part of the PTI system, normal operation of the emergency fuel system was possible.

A large portion of the test program was concerned with determining an adequate configuration for the aft fuselage since PTI operation necessarily overheated the aircraft's

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structure in that section; cooling air inlet and outlet scoops were added to the aft canted frame in addition to stiffener plates in two places on the inner side and straps in three places on the outside. It was also necessary to remove a lower portion of the aspiration as well as to add heat reflecting shields. Access doors were placed to allow servicing and adjustment of the nozzle actuator. A complete satisfactory configuration for the aft fuselage was developed and a service bulletin was formulated. One of the design objectives of the PTI systems was that it might allow installation in the field; such an objective was realized but approximately 600 man-hours would be necessary for the installation and rework operation.

The severe reduction in parts life was obviously the greatest deterrent to the use of PTI. It was estimated that it would be possible to perform six missions, utilizing 6 to 10 minutes of PTI per mission, before it would be necessary to replace turbine buckets. In addition, at least one of the PTI missions would be necessary to check-out and adjust the system; particularly critical was the determining of the proper fuel schedule for operation above an altitude of 45,000 feet. Near the end of the testing program, some work was accomplished utilizing flameholding elements in an effort to increase the life of the turbine buckets which had previously served the purpose of flameholders.

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SECTION VI

GENERAL CONCLUSIONS

Without overtemperature, no increase in thrust is made possible through overspeeding the rotor of the J47-GE-13 engine.

Immediate thrust augmentation of the J47 engine may be obtained through overtemperature operation. Such overtemperature operation requires only the decreasing of the exhaust nozzle area by placing restrictor segments in the tail-pipe. This tabbing operation is easily accomplished and requires only a small expenditure of work. Other than the decrease in the life of the engine's hot parts subjected to overtemperature, engine operation is unaffected throughout the entire operational range of the F-86 aircraft. The rate-of-climb of the F-86E aircraft may be increased by as much as four times at an altitude of 45,000 feet by overtemperaturing the J47-GE-13 engine by 18%. At lower altitudes, the increase in the rate-of-climb of the F-86 aircraft resulting from overtemperature operation will not be as pronounced. Overtemperature operation of the J47-GE-13 engine may be safely utilized for a maximum period of ten minutes before a hot parts inspection and turbine bucket replacement is necessary. Overtemperature operation of the J47-GE-27 engine in the F-86F aircraft is expected to offer a slightly lower increase in performance, relatively speaking when compared with the J47-GE-13 engine in the F-86E aircraft due to the formers initially better altitude characteristics; hot parts' life would also tend to increase slightly as the J47-GE-27 engine incorporates improvements in its hot section.

Liquid nitrogen injection into the J47 engine is not practicle for use in the F-86 aircraft. Further work with compressor refrigerant injection should be carried out on advanced engines as it would undoubtedly prove worthwhile for supersonic applications.

Many factors combine to make water-alcohol injection into the J47 engine undesirable for use in the F-86 aircraft although substantial increases in performance were demonstrated in flight tests. The necessity of having a dual tank system

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and the lack of an adequate fluid injection metering control would compromise the overall performance of the F-86 aircraft. Without further development of the water-alcohol injection system, unsatisfactory operation of the engine could be expected above an altitude of 45,000 feet. Installation of a water-alcohol injection system into the F-86 aircraft is a major task although no redesign of the aircraft structure is necessary. The J47 engine may be expected to hold up under two hours of water-alcohol injection before an inspection is necessary.

Pre-turbine fuel injection met all initial performance expectations with the exception of hot parts' life. Operation of the PTI system in the F-86 aircraft is satisfactory up to an altitude of 45,000 feet and may be considered marginally satisfactory above that to altitudes slightly in excess of 50,000 feet. The rate-of-climb of the F-86F aircraft is increased over four times at altitudes greater than 45,000 feet and the maneuverability and overall performance of the aircraft is increased substantially above an altitude of 20,000 feet. Overtemperature operation of the basic engine during pre-turbine injection alone provided nearly 40% of the total augmentation provided by PTI and although the PTI portion of the system can have a reasonably lengthy parts' life, the basic engine's hot parts would be limited to less than one hour of PTI operation. The PTI system may be installed in the field as installation criterion is available in service bulletin form, however, it might pose somewhat of a task and installation had better be accomplished at a depot. Further work on a system employing the pre-turbine injection principle is warranted although utilizing a more modern engine than the J47. Such a system offers a means for achieving a low dry loss, light weight afterburning system which could possibly find application for take-off purposes thereby not encountering the usual difficulty with afterburning at altitude and also necessitate only a minimum of complication through control requirements. Work should be directed toward improving the vaporization cooling of the turbine through better fuel distribution and the investigation of flameholders, possibly of the retractable type. Interstage turbine fuel injection on multistage engines should be further investigated. Such work by the U.S. Navy, utilizing a J46 engine, has been relatively unsuccessful to date.

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Means of increasing the performance of the F-86 aircraft were demonstrated through the program and such means would have considerably enhanced the aircraft's combat capability as may be observed by superimposing the previous discussed data upon similar data of the MiG-15 aircraft given in the appendix for comparative purposes. However, the most serious deterrent to the possible use of any of the augmentation systems brought under development in the program was the severe reduction in engine parts' life which was brought about by their use. Although the J47 engine is basically a reliable, easily maintainable engine, which can take considerable maltreatment, it is prone to turbine wheel failures and any operation which subjects the engine to overtemperature should only be resorted to in dire emergencies. Attacking the problem at its source, by providing more durable basic engine hot parts, was beyond the scope of the program.

Each phase of the project was typified by rapid progress in the beginning with further improvements requiring intensified effort and considerable time as each system became more complicated and increased altitude requirements demanded more sophistication. Refinement of each system soon reached a saturation point where returns were diminishing due to the inherent limitations of the basic engine and the high density of the aircraft's structure.

It is obvious that each of the thrust augmentation means developed for the J47 engine in the F-86 aircraft was of a strictly war emergency type and only suitable for use in an extreme emergency and then only when adequate logistic and maintenance support can be made available or where flight safety or the aircraft in-commission rate can be compromised for the increased performance gains.

The thrust augmentation program for the J47 engine in the F-86 aircraft demonstrated that there is no substitute for a basically better engine which is made available through a normal development program. This point was illustrated in the superior performance and combat record of the F-86F aircraft which uses the J47-GE-27 engine as compared to the F86E which is powered by the J47-GE-13 engine. The J47-GE-27 engine is a development of the J47-GE-13 engine.

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REFERENCES

The project's official engineering file, which was maintained by the Power Plant Laboratory, Wright Air Development Center served as the primary reference for WADC TN 55-298 in addition to information gathered from personnel who participated in the program. Aside from the project's record books, correspondence, minutes, and memorandums, both Government' and Contractors' reports were maintained in the file and the more significant reports which were used as references are given below.

Air Force Flight Test Center. Altitude Thrust Augmentation Using Water-Alcohol Injection. AF Technical Report No. AFFTC 53-8. March 1953. (Confidential Report)

Air Force Flight Test Center. Phase II Performance And Serviceability Tests Of The F-86F Airplane USAF No. 51-13506 With Pre-Turbine Modifications. AF Technical Report No. AFFTC 54-16. June 1954. (Confidential Report)

General Electric Company. Flight Test Of Overtemperature War Emergency Power Setting On A J47-GE-13 Engine Installed In A F-86e Aircraft At AFFTC Edwards, California. GE Bulletin No. DF52GT745. 23 June 1952. (Unclassified Report)

General Electric Company. Estimated Minimum Performance Of The General Electric J47-GE-27 Turbojet Engine At War Emergency Rating With Preturbine Injection Kit. GE Bulletin No. R53AGT568. 15 September 1953. (Confidential Report)

General Electric Company. Model Specification War Emergency Rating System Kit No. 7032R37. GE Specification No. E-641. 31 December 1953. (Confidential Specification)

General Electric Company. Pre-Turbine Fuel Injection for Thrust Augmentation. GE Technical Information Series No. R55AGT95. 11 March 1955. (Confidential Report)

North American Aviation, Inc. Thrust Augmentation Of A J47-GE-15 Engine By Means Of Liquid Nitrogen Injection Into The Compressor Inlet. NAA Report No. NA52-297. 19 June 1952. (Confidential Report)

North American Aviation, Inc. F-86F Airplane With Pre-Turbine Injection (PTI) Thrust Augmentation. NAA Summary Report No. NA-54-664. 9 July 1954. (Confidential Report)

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National Advisory Committee For Aeronautics. (Confidential Title) Altitude Investigation Of Thrust Augmentation Of A J47-GE-27 Turbojet Engine By Injecting Additional Fuel Immediately Ahead Of The Turbine. NACA Research Memorandum RM E53L31, 18 December 1953. (Confidential Report)

Air Technical Intelligence. (Secret Title). Technical Report TR-AC-27. 13 October 1953. (Secret Report)
As the title of this report is Secret it can not be given here. The material taken from the above report and contained in WADC TN-55-298 is as of the writing of this TN classified Confidential.

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APPENDIX

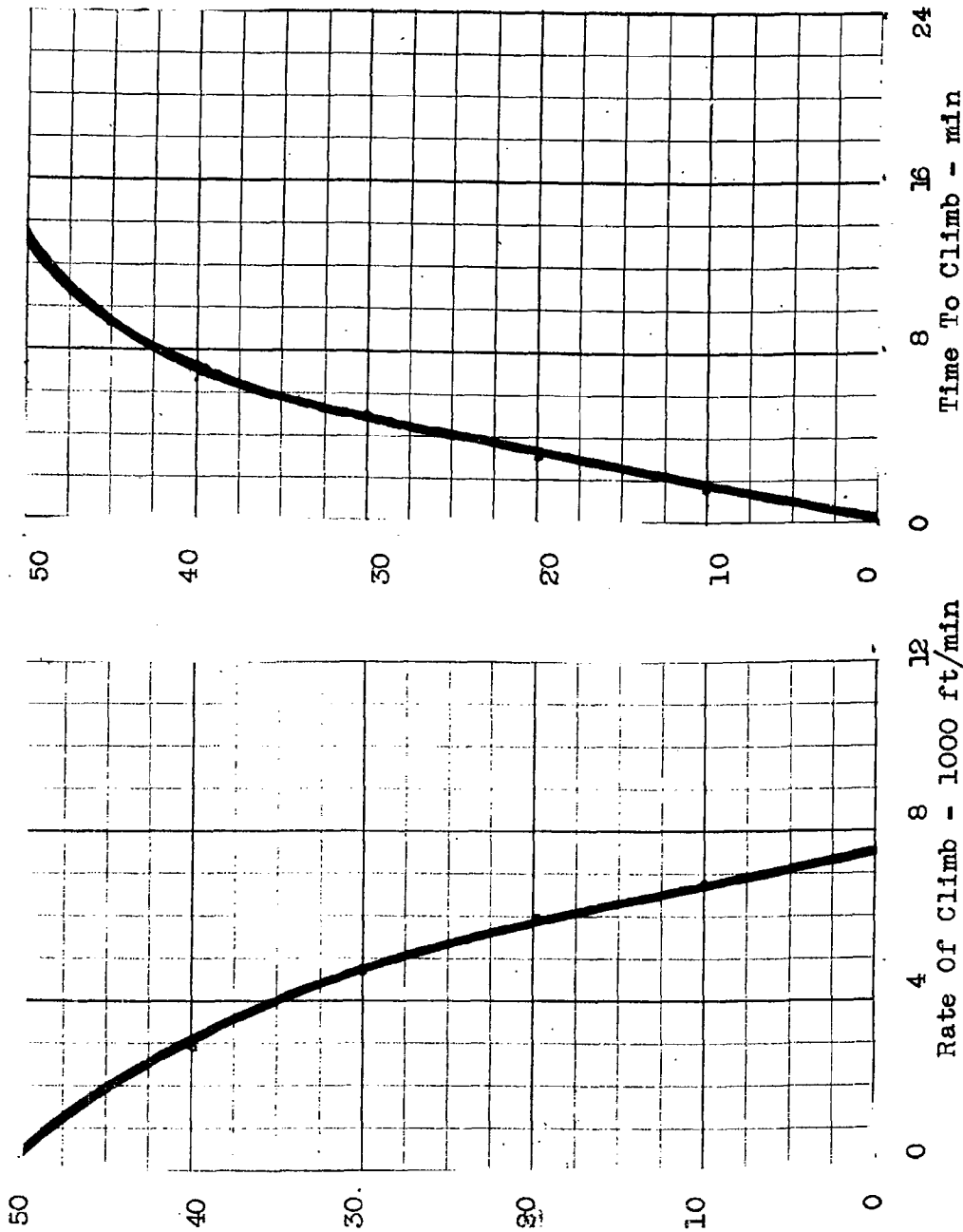


FIGURE 35. MiG-15 Aircraft. Combat Weight. Clean.

FIGURE 36. MiG-15 Aircraft. Combat Weight. Clean.

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Armed Services Technical Information Agency

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DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE OHIO

FEB 19 2002

MEMORANDUM FOR DTIC/OCQ (ZENA ROGERS)
8725 JOHN J. KINGMAN ROAD, SUITE 0944
FORT BELVOIR VA 22060-6218

FROM: AFMC CSO/SCOC
4225 Logistics Avenue, Room S132
Wright-Patterson AFB OH 45433-5714

SUBJECT: Technical Reports Cleared for Public Release

References: (a) HQ AFMC/PAX Memo, 26 Nov 01, Security and Policy Review,
AFMC 01-242 (Atch 1)

(b) HQ AFMC/PAX Memo, 19 Dec 01, Security and Policy Review,
AFMC 01-275 (Atch 2)

→ (c) HQ AFMC/PAX Memo, 17 Jan 02, Security and Policy Review,
AFMC 02-005 (Atch 3)

1. Technical reports submitted in the attached references listed above are cleared for public release in accordance with AFI 35-101, 26 Jul 01, *Public Affairs Policies and Procedures*, Chapter 15 (Cases AFMC 01-242, AFMC 01-275, & AFMC 02-005).

2. Please direct further questions to Lezora U. Nobles, AFMC CSO/SCOC, DSN 787-8583.

LEZORA U. NOBLES
AFMC STINFO Assistant
Directorate of Communications and Information

Attachments:

1. HQ AFMC/PAX Memo, 26 Nov 01
2. HQ AFMC/PAX Memo, 19 Dec 01
3. HQ AFMC/PAX Memo, 17 Jan 02

cc:
HQ AFMC/HO (Dr. William Elliott)



DEPARTMENT OF THE AIR FORCE

HEADQUARTERS AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE OHIO

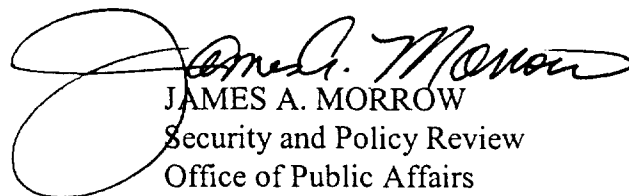
JAN 17 2002

MEMORANDUM FOR HQ AFMC/HO

FROM: HQ AFMC/PAX

SUBJECT: Security and Policy Review, AFMC 02-005

1. The reports listed in your attached letter were submitted for security and policy review IAW AFI 35-101, Chapter 15. They have been cleared for public release.
2. If you have any questions, please call me at 77828. Thanks.


JAMES A. MORROW
Security and Policy Review
Office of Public Affairs

Attachment:
Your Ltr 14 January 2002

14 January 2002

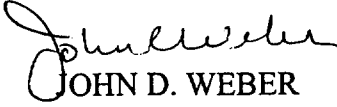
MEMORANDUM FOR: HQ AFMC/PAX
Attn: Jim Morrow

FROM: HQ AFMC/HO

SUBJECT: Releasability Reviews

1. Please conduct public releasability reviews for the following attached Defense Technical Information Center (DTIC) reports:
 - a. *Flight Test Program for Model P-86 Airplane Class – Jet Propelled Fighter*, 2 December 1946; DTIC No. AD-B804 069.
 - b. *Physiological Recognition of Strain in Flying Personnel: Eosinopenia in F-86 Combat Operations*, September 1953; DTIC No. AD- 020 375.
 - c. *Phase IV Performance Test of the F-86F-40 Airplane Equipped with 6x3-inch Leading Edge Slats and 12-inch Extensions on the Wing Tips*, May 1956; DTIC No. AD- 096 084.
 - d. *F-86E Thrust Augmentation Evaluation*, March 1957; DTIC No. AD- 118 703.
 - e. *F-86E Thrust Augmentation Evaluation*, Appendix IV, March 1957; DTIC No. AD- 118 707.
 - f. *A Means of Comparing Fighter Effectiveness in the Approach Phase*, October 1949; DTIC No. AD- 223 596.
 - g. *War Emergency Thrust Augmentation for the J47 Engine in the F-86 Aircraft*, August 1955; DTIC No. AD- 095 757.
 - h. *Operational Suitability Test of the F-86F Airplane*, 4 May 1953; DTIC No. AD- 017 568.
 - i. *Estimated Aerodynamic Characteristics for Design of the F-86E Airplane*, 26 December 1950; DTIC No. AD- 069 271.
 - j. *Combat Suitability Test of F-86F-2 Aircraft with T-160 Guns*, August 1953; DTIC No. AD- 019 725.

2. These attachments have been requested by Dr. Kenneth P. Werrell, a private researcher.
3. The AFMC/HO point of contact for these reviews is Dr. William Elliott, who may be reached at extension 77476.


JOHN D. WEBER
Command Historian

10 Attachments:

- a. DTIC No. AD-B804 069
- b. DTIC No. AD- 020 375
- c. DTIC No. AD- 096 084
- d. DTIC No. AD- 118 703
- e. DTIC No. AD- 118 707
- f. DTIC No. AD- 223 596
- g. DTIC No. AD- 095 757
- h. DTIC No. AD- 017 568
- i. DTIC No. AD- 069 271
- j. DTIC No. AD- 019 725